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USAAVLABS TECHNICAL NOTE 8

FLIGHT TEST EVALUATION OF THE THREE-AXIS MECHANICAL STABILITY AUGMENTATION SYSTEM

By
George W. Fosdick

December 1970

EUSTIS DIRECTORATE
U. S. ARMY AIR MOBILITY RESEARCH AND DEVELOPMENT LABORATORY
FORT EUSTIS, VIRGINIA

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USAAVLABS Technical Note 8
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SUMMARY

This note presents the results of flight tests conducted to evaluate a three-axis mechanical stability augmentation system (MSAS), known as "Dynagyro", on a UH-1 helicopter. The purpose of a stability augmentation system is to augment the stability and control characteristics of unstable or weakly stable aircraft so as to provide satisfactory flying qualities. The tests encompassed 9-1/2 flight hours and approximately 3 hours of ground and hangar testing. The MSAS included an entirely new concept: vortex valve fluidic servoactuators.

The magnitude of the installation and conversion procedure for installing the MSAS was relatively small.

The MSAS, as tested, did not perform as well in the helicopter as it did in the laboratory or as well as theory indicated it should.

The MSAS did not require a heat exchanger for fluid temperature control.

The engagement and disengagement transients were acceptable. Helicopter response was decreased significantly following small-amplitude step displacement of the flight controls. The MSAS was ineffective in improving lateral-directional damping.

The yaw SAS responded properly during autorotational entry; however, it functioned improperly during the remaining tests. Pilot acceptance of the MSAS was poor. The magnitude of the installation and conversion procedure for the test MSAS is not representative of the procedure for a production MSAS. The MSAS was not compatible with the operating environment of the UH-1H helicopter.

The improper functioning of the yaw SAS contributed to the ineffectiveness of the MSAS in improving lateral-directional damping. The yaw SAS provides insignificant yaw damping during autorotational entry. Improper MSAS functioning and helicopter attitude limitations contributed to the poor pilot acceptance of the MSAS.

FOREWORD

The flight test evaluation was conducted to obtain dynamic stability and control response data for a helicopter with the MSAS installed; to obtain a qualitative evaluation of the MSAS by several helicopter pilots; and to determine the magnitude of the installation and conversion procedures of the MSAS in a UH-1 helicopter. This work was performed by the Eustis Directorate, U. S. Army Air Mobility Research and Development Laboratory, Fort Eustis, Virginia, under House Task AM 68-6, during the period from February 1968 through June 1969.

The following USAAVLABS personnel contributed to this program:

Mr. R. P. Smith	Lead Engineer, S&C Branch
Mr. G. W. Fosdick	Project Engineer
Mr. W. D. Vann	Aerospace Engineer
Mr. B. J. Jones	Engineering Technician
Mr. A. M. Williamson	Electronic Technician
Mr. D. R. Etter	Electronic Technician
Mr. J. M. Hayth	Engineering Technician

The following U. S. Army Aviation Systems Test Activity (USAASTA) personnel contributed to this program and to this report:

LTC Dennis M. Boyle	Director of Flight Test
Maj Wayne B. Davis	Experimental Test Pilot

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INTRODUCTION

All helicopters are inherently unstable in some speed regimes. Modern design practices have brought the degree of instability within the pilot's control capability, but flight safety and pilot fatigue usually dictate the need for some sort of stability augmentation. Acceptable flying characteristics can be provided through the use of either mechanical or electronic stabilization systems. Although both types of these systems are presently in common use, they have marked advantages and disadvantages.

Electronic stabilization systems can be very light in weight. Also, the inherent flexibility of electronic circuitry permits the designer to closely tailor the system's characteristics to the requirements of the helicopter or, if desired, to provide for the execution of preprogrammed maneuvers. On the other hand, the electronic system is highly complex and costly to produce, requires extensive maintenance by highly skilled personnel, and is relatively low in reliability.

Mechanical stabilization systems of the type used on some helicopters rely on a gyroscope to sense the attitude deviation rate of the helicopter and to provide stabilizing signal inputs to the helicopter control system. The gyroscope motion is damped by viscous or aerodynamic dampers which provide a restoring torque that continuously seeks to align the gyroscope axis with a fixed reference axis in the helicopter. The mechanical stabilization systems in current use require very little maintenance and are highly reliable, but they are heavy as compared to electronic systems.

It would, of course, be desirable to provide a stabilizing system that possesses the lightweight characteristics of current electronic systems and the high reliability characteristics of current mechanical systems. To achieve this, attempts have been made to reduce the size of the mechanical system. Previous attempts have not been fruitful for a two-degree-of-freedom system because of the problems encountered in providing either viscous or aerodynamic dampers that possess damping characteristics compatible with the requirements of miniaturized mechanical gyros. Hence, the damper problem has been the primary stumbling block in attempts to miniaturize mechanical stabilizers.

The Dynasciences Corporation recently developed a method for the application of coulomb damping to the damping of gyroscopes. This method of damping was applied to a stability augmentation device for helicopters, resulting in the mechanical stability augmentation system (MSAS).

The MSAS, delivered to USAAVLABS by Dynasciences, was completely overhauled by USAAVLABS personnel, and a flight test evaluation installation for a UH-1D helicopter was designed and fabricated. The installation consisted of the hydraulic power supply as well as the hydraulic and electrical power supply control system. An extensive laboratory operational test of the complete installation was then carried out at USAAVLABS.

The flight test evaluation was performed at the U. S. Army Aviation Systems Test Activity, Edwards AFB, California. The objectives of the evaluation were:

1. Determination of the magnitude of the installation and conversion procedure.

2. Quantitative measurement of the system's performance.

3. Qualitative evaluation of the system by at least three pilots.

4. Determination of any changes necessary to enhance the system's performance and serviceability.

The objectives of the evaluation were met during eight flights, totaling 9-1/2 flight hours.

DESCRIPTION OF THE TEST HARDWARE

The test helicopter, Figure 1, was a YUH-1H 60-6033. The helicopter was modified only in those areas required to accept the mechanical stability augmentation system installation and the instrumentation.

The mechanical stability augmentation system tested consisted of a coulomb-damped two-degree-of-freedom gyroscope (Dynagyro) and a single-axis spring-damped rate gyroscope (Heading Assist).

A detailed description of the Dynagyro is presented in Reference 1.

The Heading Assist gyroscope is hydraulically powered. The major components of the drive system consist of a planetary gear transmission and a universal joint. The Heading Assist gyro senses the change in aircraft directional angular rate rather than the change in aircraft angular displacement. The signal is integrated into the aircraft control system, through a control boost actuator and mixing linkages, in a manner similar to the Dynagyro.

In order to introduce the gyro control input into the helicopter control system, it is necessary to reduce the pilot's control to the swash plate such that with the integrated system the sum of pilot and gyro input motion equals the maximum pilot input prior to integration.

The MSAS specifications are summarized in Table I.

The MSAS, as packaged for installation in the test helicopter, is shown in Figures 2 through 9.

The installation included a self-contained high-pressure (1500-psi) hydraulic power supply (Figures 10, 11, and 12) for driving the gyros as well as the servoactuators.

Details of the fluidic servoactuators are provided in Reference 2. The servoactuator authority in the longitudinal and the lateral control system was ± 15 percent of the total control travel; in the directional control system, ± 25 percent of the total control travel.

TABLE I. MSAS SPECIFICATIONS

Item	Dynagyro	Heading Assist Gyro
Weight	16.3 lb	14.5 lb
Size	9.5 in. x 9.5 in. x 12.5 in.	6.5 in. x 8.25 in. x 12.5 in.
Speed	4000 rpm	9700 rpm
Angular Momentum	92 in.-lb-sec	100 in.-lb-sec
Damping	0.008 rad/sec	175 in.-lb/rad/sec
Spring Rate	-	50 in.-lb-rad
Fluid Flow	1.12 gpm	0.62 gpm
Pressure	1500 psi	1500 psi
Power Requirement	1.0 hp	0.54 hp

FABRICATION AND INSTALLATION

FABRICATION AND CHECKOUT

The flight test package, consisting of supporting structure and inclosures for the Dynagyro, the Heading Assist gyro, and the fluidic servoactuators, was fabricated and assembled (Figures 13, 14, and 15). The hydraulic power supply package and control console are shown in Figures 3 and 16. The instrumentation package is shown in Figures 6, 8, and 9.

Laboratory testing and checkout of the complete MSAS package consisted of mounting the MSAS on a three-degree-of-freedom tilt table and operating as an integral self-contained system (Figures 13 and 14). During these tests, gyro and servoactuator performance was checked by simulating aircraft upset by use of the tilt table. Gyro rpm was also checked, as well as the operating temperature of the hydraulic pump and motor, the heading assist step-up transmission, and the oil temperature into the pump, the servoactuators, and the reservoir.

INSTALLATION AND CHECKOUT

The MSAS and instrumentation were installed in the YUH-1H at USAASTA. The Dynagyro module and the Heading Assist module were mounted on the cabin floor just forward of the pylon structure (Figure 6). The hydraulic power supply system was mounted on the cabin floor just to the left of the pylon structure. The instrumentation package was mounted on the cabin floor just to the right of the pylon structure (Figures 6, 8, and 9).

After the installation was completed, the safety-of-flight review board convened, and the following limitations were written into the USAAVLABS safety-of-flight release: (1) MSAS cutoff switch must be installed on the pilot's cyclic control stick; (2) the test flights must be flown with a pilot and a copilot at the controls; (3) surface wind must be less than 5 knots for hovering flight in cross-wind, down-wind, and confined areas; and (4) helicopter takeoff gross weight must be less than 8000 pounds.

The magnitude of the installation and conversion procedure of the helicopter for installing the MSAS was relatively small due to the simplicity of the system, the minimum interfaces between the package and the aircraft systems, and the method used to mount the package onto the heli- ⍴ter floor without any structural modification requirements.

The test helicopter was weighed after the MSAS and flight test instrumentation were installed; the gross weight was 7700 pounds, with the center of gravity located at station 137.

TEST PROCEDURES AND RESULTS

GROUND TEST

Following the installation of the MSAS and the flight test instrumentation, all systems were operationally checked with the helicopter on the ground.

The operational checkout consisted of vibratory frequency sweeps of the pilot controls with the MSAS shut off and turned on. The system was also cycled on and off by removing and applying both electrical power and hydraulic power.

Test conditions that were checked included longitudinal, lateral, and directional control pulse and step inputs at several pilot-selected engine power settings, and MSAS engagements and disengagements by all methods available. During these tests, application of rapid forward or aft control pulses resulted in a high-frequency self-sustaining vibration in the helicopter control system, as well as in the MSAS longitudinal servoactuator. After extensive investigation, the high-frequency vibration was determined to be a hydraulic resonance condition within the fluidic servoactuator. The servoactuators were modified, by the manufacturer, to eliminate the resonance problem. The modified servoactuators were reinstalled in the MSAS, and the entire ground operation control input sequence was completed satisfactorily.

FLIGHT TEST

The same basic tests were used for each flight throughout the flight evaluation. A flight-test log for the MSAS is presented in Table II. The flight-test card items consisted of pulse, step, doublet, and dutch roll excitation control inputs while the helicopter was stabilized in hovering flight, at level flight speeds of 60, 90, and 110 KIAS.

TABLE II. MECHANICAL STABILITY AUGMENTATION SYSTEM FLIGHT TEST LOG

Date	Event	Results
5 March 1969	MSAS and instrumentation shipped to USAASTA from USAAVLABS	
11 March 1969	Completed shipment of MSAS and instrumentation	
24 March 1969	Flight-safety team review	Released with specific limitations and requirements.
28 March 1969	Aircraft functional ground run check - 45 minutes	High-frequency oscillation in flight control system after longitudinal pulse control input.
1 April 1969	Received safety-of-flight release from USAAVLABS	
2 April 1969	Fluidic servoactuators returned to manufacturer for modification	Modified to eliminate the high-frequency oscillation.
9 April 1969	Fluidic servoactuators reinstalled on MSAS	
15 April 1969	Aircraft functional ground run check - 30 minutes	Completed flight control input sequence satisfactorily. Hovered for 2 minutes when abrupt high-frequency noise started from behind the cabin.
17 April 1969	Aircraft maintenance test flight	Excessive EGT at idle RPM - determined to be caused by FOD; engine change required.
19 April 1969	Flight test team returned to USAAVLABS	
5 May 1969	Flight test team resumed activities at USAASTA	Maintenance test flight.
8 May 1969	1-1/2-hour flight test	Configuration 1 forward flight data.
9 May 1969	1-1/2-hour flight test	Configuration 1 hovering flight data.

TABLE II. CONTINUED

Date	Event	Results
12 May 1969	50-minute flight test	Configuration 2 level flight data. Yaw SAS gyro centering spring attachment bolt broke.
12 May 1969	30-minute flight test	Yaw SAS gyro centering spring attachment bolt broke as soon as testing started.
12 May 1969	Postflight revealed yaw SAS gyro balancing plug missing	Flights discontinued until repairs completed.
13 May 1969	1-1/4-hour flight test	Configuration 1 with gyros operating. Dynagyro anti-tumbling mechanism did not function satisfactorily.
14 May 1969	Dynasciences personnel on site to eliminate MSAS ills	
19 May 1969	1-hour flight test	Configuration 1 with gyros operating.
19 May 1969	Removed helicopter stabilizer bar	
20 May 1969	1-1/2-hour flight test	Configurations 3 and 5 level flight data points.
20 May 1969	1-1/4-hour flight test	Configurations 3 and 5 hovering flight data points.
20 May 1969	Flight test evaluation concluded	Total flying time 9 hours 25 minutes.

FLIGHT TEST RESULTS

The MSAS, as tested, did not perform as well in the helicopter as theory indicated it should. A portion of this poor performance may be attributed to the use of a UH-1H helicopter. The theoretical and laboratory work on this program used the UH-1B helicopter and its performance data as a basis for establishing MSAS requirements. The UH-1H helicopter was used because of the nonavailability of a UH-1B helicopter. The MSAS, as packaged, did not require a heat exchanger to control fluid temperature within the desired tolerances. The engagement and disengagement transients were well within acceptable limits. The helicopter response following a small-amplitude step displacement of the flight controls was decreased significantly, in comparison to the basic helicopter response. The simple "hardware concept" system did not provide a pilot loop; it was a compromise between being optimum for hover and for cruise-speed level flight.

Data obtained from flight tests of the MSAS installed in the UH-1H helicopter were read, and plots of the pertinent data were prepared. The procedures and techniques used are summarized below. Appendix I contains a representative sample of the data obtained during the flight test program (Right Lateral Pulse Input).

Data Processing

For each flight test condition, the output data for the various control inputs were the helicopter angular attitudes in pitch, roll, and yaw and the helicopter angular rates in pitch, roll, and yaw. Each output trace was read for all of the flight conditions to obtain the helicopter attitude after 1 second and the helicopter angular velocity after 1 second. All readings were taken at 0.1-second intervals to obtain the peak angular acceleration of the helicopter. The traces were read at the midpoint of any oscillations that were present. Each helicopter angular velocity trace was read for the first several output cycles to be sure of obtaining the peak acceleration. The helicopter angular acceleration was computed using a numerical differentiation of the angular velocity trace for each output axis.

Damping Ratio Computation

A damping ratio was computed using the techniques of AFFTC-TN-59-21, Air Force Flight Test Center Stability and Control Techniques. Only one of the output quantities was used to obtain the damping ratio. The helicopter angular rates were used for this purpose with a primary output selected for each test condition. This selection was based on the nature of the input. That is, for a lateral control input, the helicopter roll rate was taken as the primary output; for a directional control input, the helicopter yaw rate was taken as the primary output. The damping ratio is based on the response of a lightly damped system as it appears in the response relation:

$$X = A' \exp(-\xi \omega_n t) \cos(\omega_n \sqrt{1-\xi^2}\, t - \alpha)$$

where X = output

A' = coefficient

ξ = damping ratio

ω_n = undamped natural frequency

t = time

α = phase angle

The technique of determining ξ from the recorded data involves plotting successive peak values of the output. In order to obtain a valid slope, the output must have at least two successive readable peaks of the same sign. This was not available from all of the test conditions. Such conditions have damping that corresponds to a heavily damped system and can not be analyzed with the lightly damped relations. AFFTC-TN-59-21 treats such a case by saying that if the damping ratio is above about 0.4, the results of graphical analysis are quite inaccurate and other means of analysis should be used. As no other data were available, the cases where this was true are left blank in the damping ratio plot. In general, such situations are well within acceptable stability characteristics.

Plots

The data plots were: Airspeed (knots) versus

Peak Angular Acceleration (deg/sec^2)

Damping Ratio

Control Effectiveness - Attitude at 1.0 sec (deg)

Angular Velocity at 1.0 sec (deg/sec)

Configurations

The helicopter configurations used were:

1 - Stabilizer bar installed on the helicopter, MSAS not operating.

3 - Stabilizer bar not installed on the helicopter, MSAS not operating.

5 - Stabilizer bar not installed on the helicopter, lateral and longitudinal axis of the MSAS operating; yaw axis of the MSAS not operating.

Longitudinal Control Response

Figures 17, 18, and 19 show the results of an aft 1-inch pulse control input, configurations 1, 3, and 5 respectively. The forward pulse as well as the aft pulse produced insignificant response differences between the helicopter configurations. As would be expected, the control effectiveness at hover conditions for configuration 3 (no bar, no SAS) was noticeably greater than for the forward flight conditions.

Lateral Control Response

Figures 20, 21, and 22 show the results of a left lateral 1-inch pulse control input, and Figures 23, 24, and 25 show the results of a right lateral 1-inch pulse control input, for configurations 1, 3, and 5 respectively. The control effectiveness was generally lower for the undamped configuration than for either of the damped configurations. The damping ratio shows an increase with airspeed for the left lateral input and a decrease with airspeed for the right lateral input. This can be partially attributed to the necessity of a compromise design of the Dynagyro, i.e., damping required at hover versus damping required at high forward speed. Also, the control effectiveness of the undamped helicopter is not appreciably different from that of the damped helicopter.

Directional Control Response

The yaw SAS malfunctioned early in the flight test program; therefore, very little data were obtained. These data cannot be considered reliable.

Maneuvering Flight

The control input response tests in maneuvering flight were for configuration 1 only (stabilizer bar installed on the helicopter, MSAS not operating). The limitations that the Dynagyro imposed upon the allowable pitch and roll angles of the helicopter prohibited the execution of the maneuvers that would have provided meaningful data on damping provided by the MSAS during maneuvering flight. The MSAS was limited to ± 20° lateral and longitudinal tilt angles.

Autorotational Entry

The insignificant quantity of yaw SAS data obtained prior to yaw SAS malfunction indicated that the yaw SAS responded properly during autorotational entry to assist the pilot with yaw control of the helicopter. The maneuvers performed were of the throttle chop variety - holding the flight controls fixed as long as possible after the throttle chop, rather than immediately initiating the corrective autorotational entry control inputs. The imposed helicopter attitude limitations precluded the performance of additional throttle chop maneuvers. During autorotational entry maneuvering, the yaw SAS did not provide any apparent significant performance improvement over the performance of the helicopter without the yaw SAS operating. Samples of the autorotational oscillograph traces are included in Appendix I.

Qualitative Evaluation

The qualitative evaluation report generated by this program is included in Appendix II.

CONCLUSIONS

As a result of the test program described in this note, it is concluded that:

1. The magnitude of the installation and conversion procedure for the test MSAS cannot be considered as representative of the magnitude of the installation and conversion procedure for a possible production version of the MSAS.

2. The test MSAS is not compatible with the operational or vibrational environment of the UH-1H helicopter.

3. Adjustable damping rate provisions are required in test stability augmentation systems.

4. Hydraulic fluid temperature control provisions are required for hydraulically powered mechanical stability augmentation systems.

5. The test MSAS attitude limitations, ± 20° in pitch and roll, restricted helicopter maneuvering.

6. Helicopter response following small-amplitude flight control inputs was not satisfactory.

RECOMMENDATIONS

It is recommended that:

1. The MSAS be extensively and completely laboratory tested within a helicopter environment prior to any additional flight testing.

2. Adjustable provisions, to allow "compromise" damping rate settings, be incorporated in test stability augmentation systems.

3. Hydraulic fluid temperature control provisions be incorporated in hydraulically powered mechanical stability augmentation systems.

4. The operating range of the mechanical stability augmentation system be increased to an acceptable level for normal helicopter maneuvering.

5. A pilot loop be provided in mechanical stability augmentation systems to provide satisfactory helicopter response following small-amplitude flight control inputs.

Figure 1. Flight Test Helicopter - YUH-1H.

Figure 2. MSAS Power Supply Module.

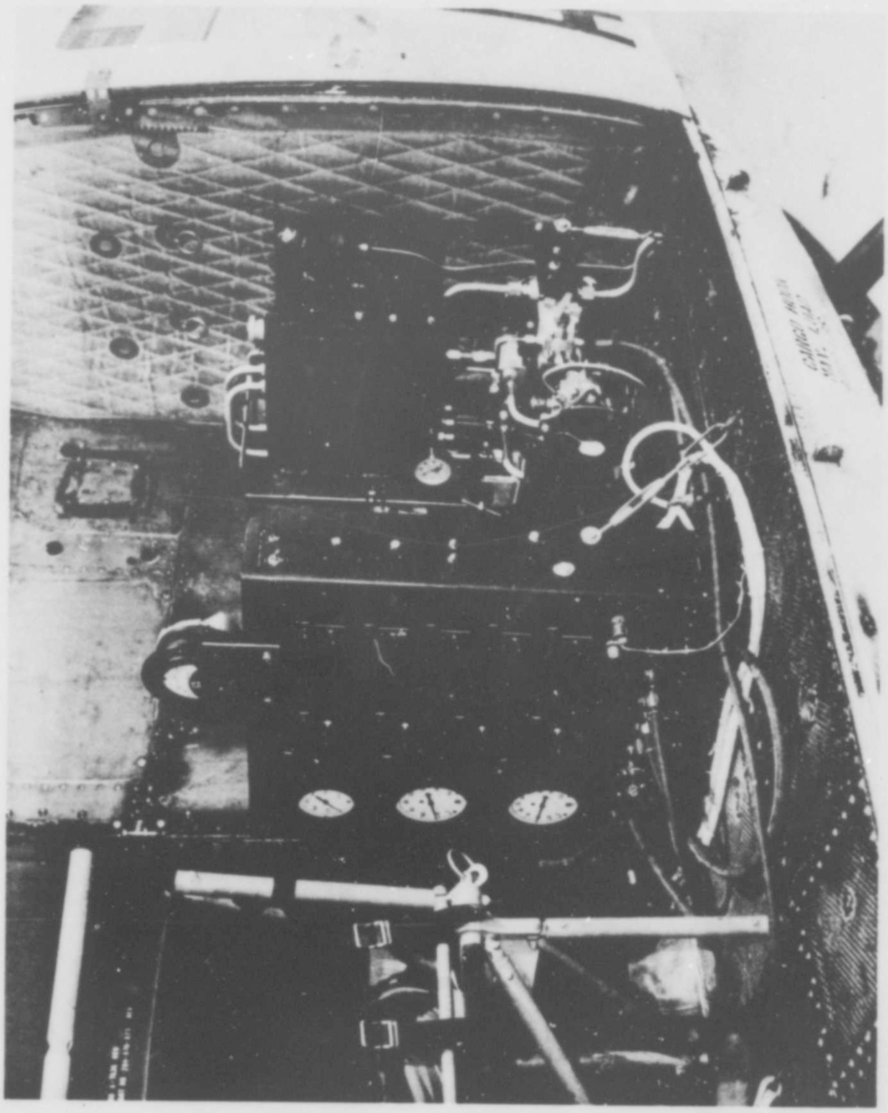


Figure 3. MSAS Power Supply Module.

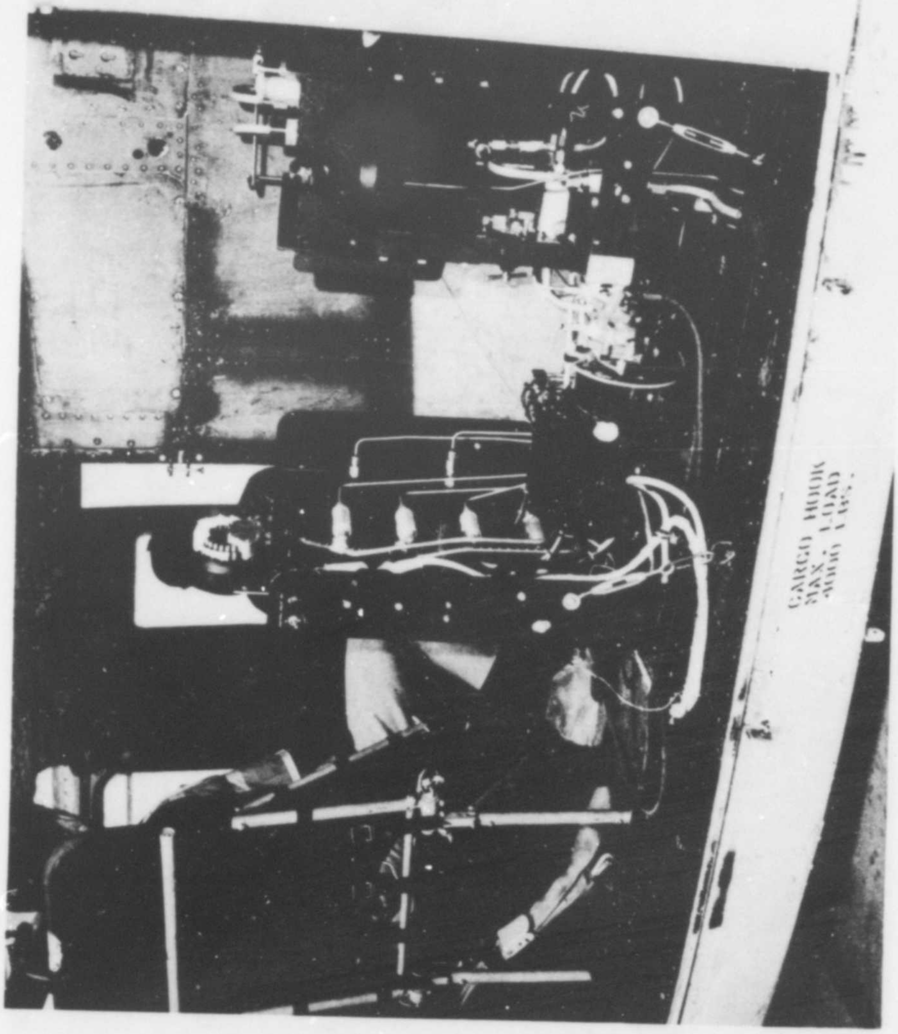


Figure 4. MSAS Power Supply Module.

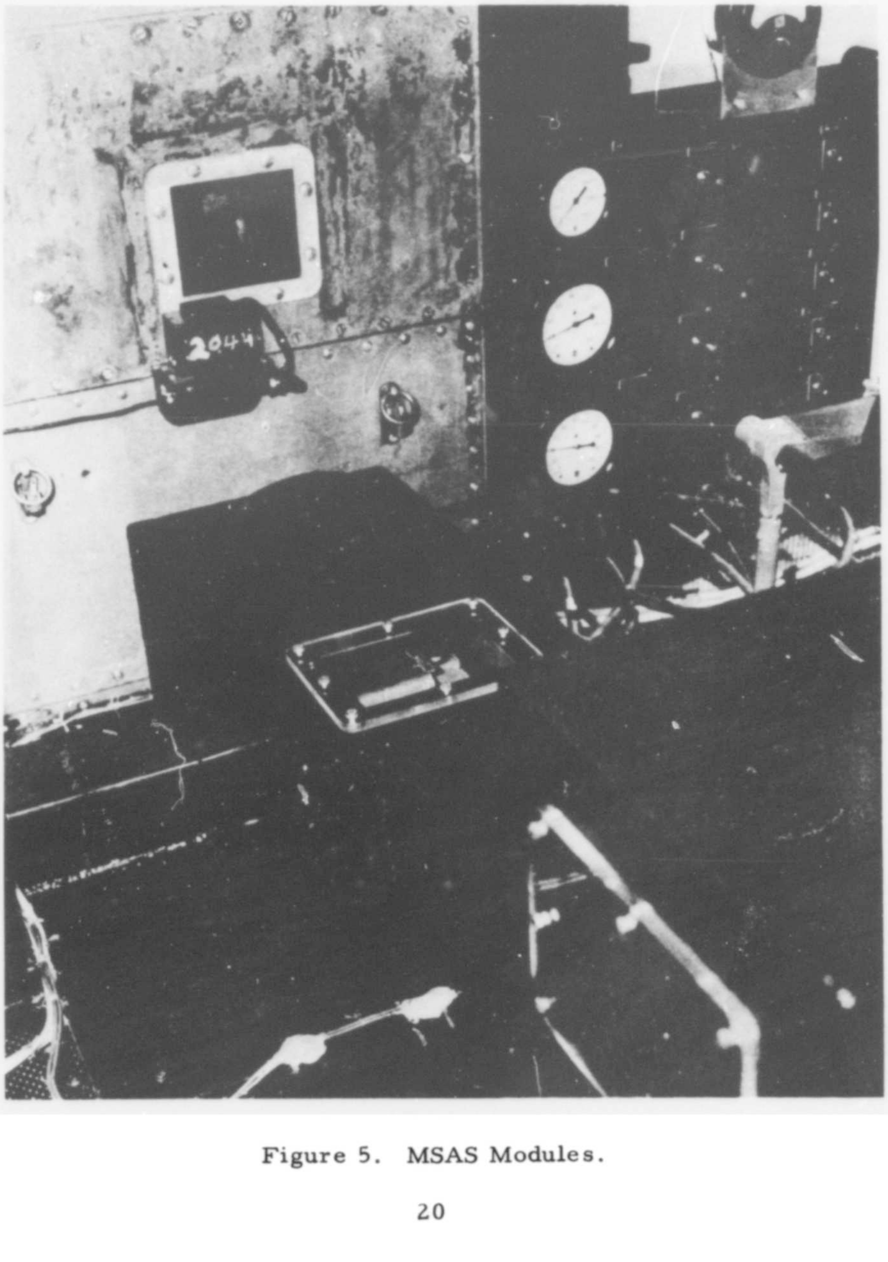


Figure 5. MSAS Modules.

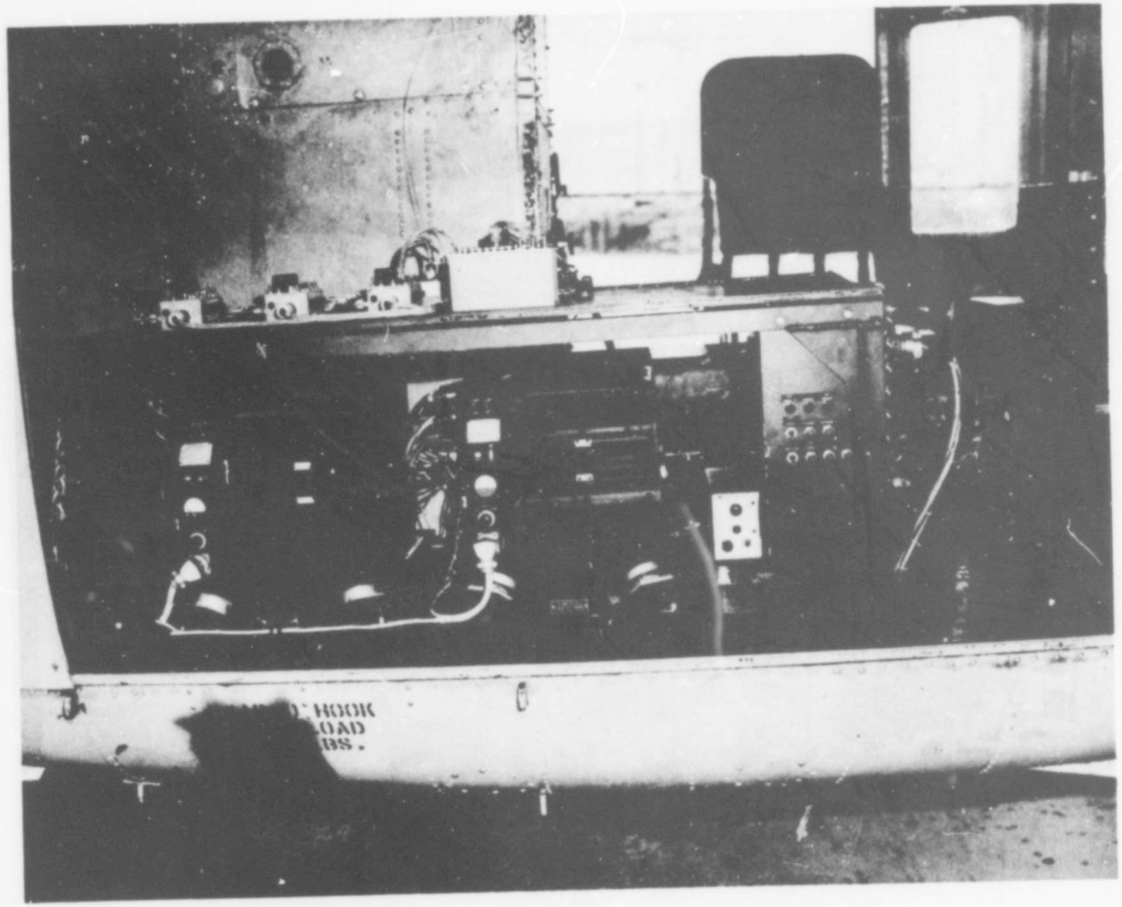


Figure 6. Instrumentation Package.

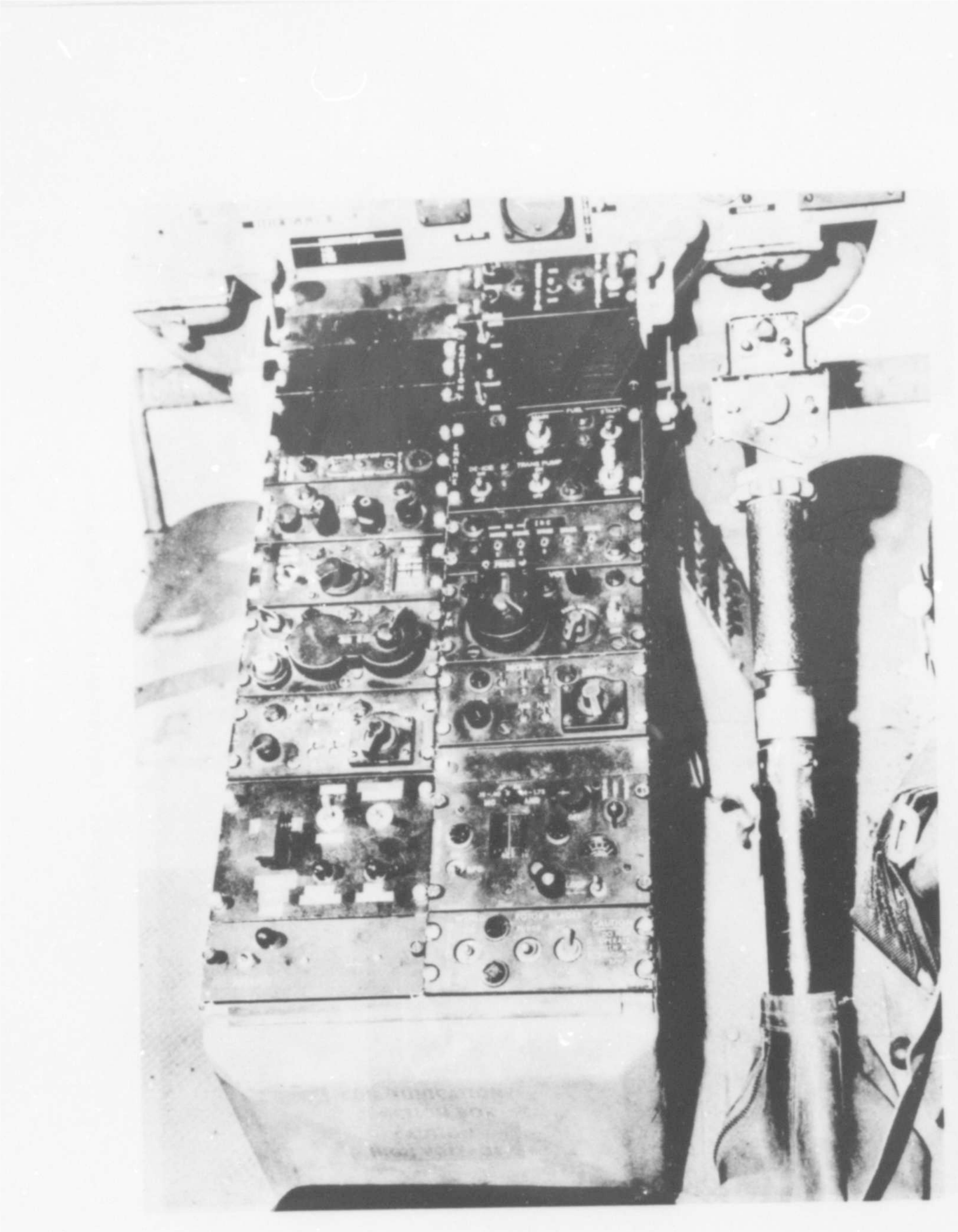


Figure 7. Pilot's Console.

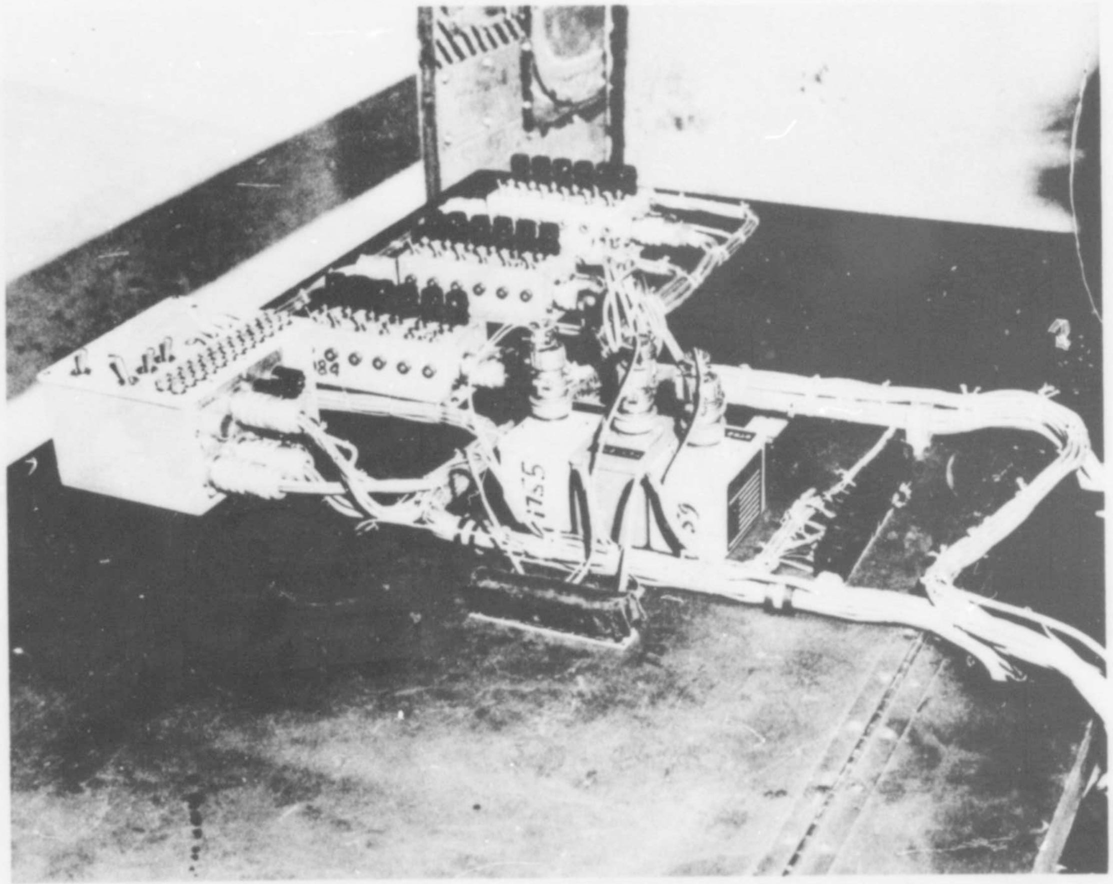


Figure 8. Instrumentation Package.

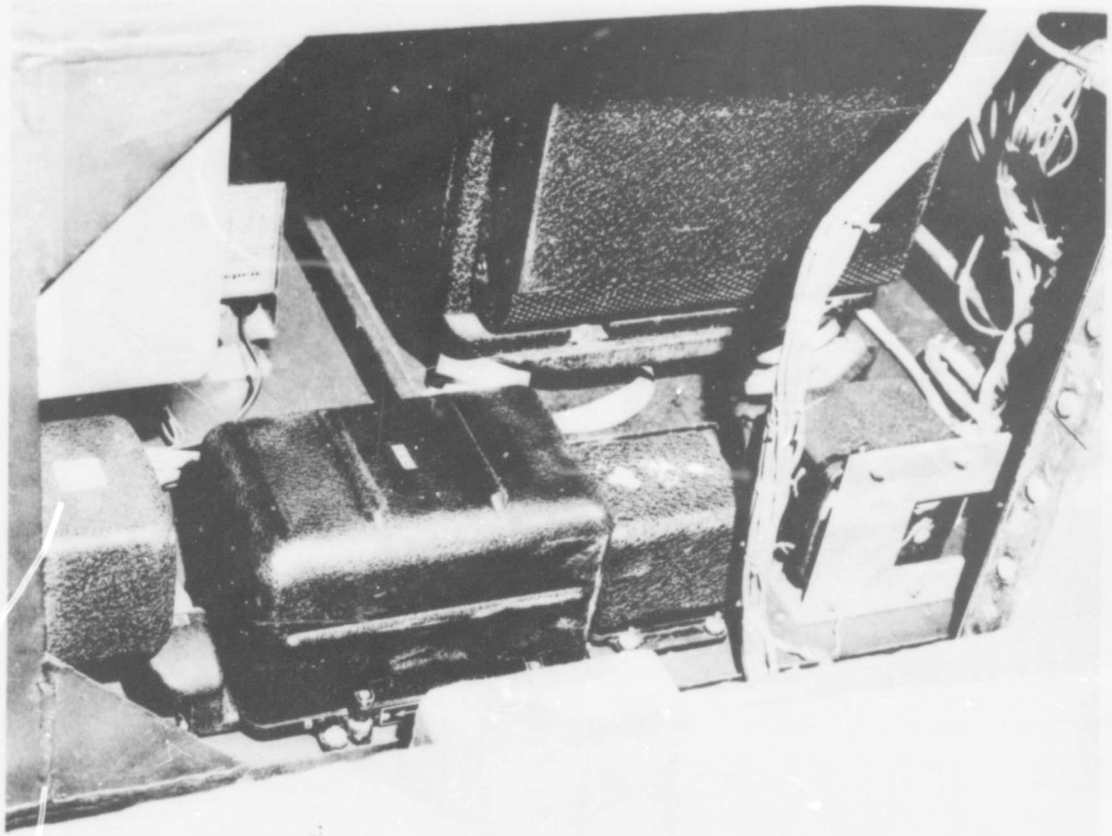


Figure 9. Instrumentation Gyros.

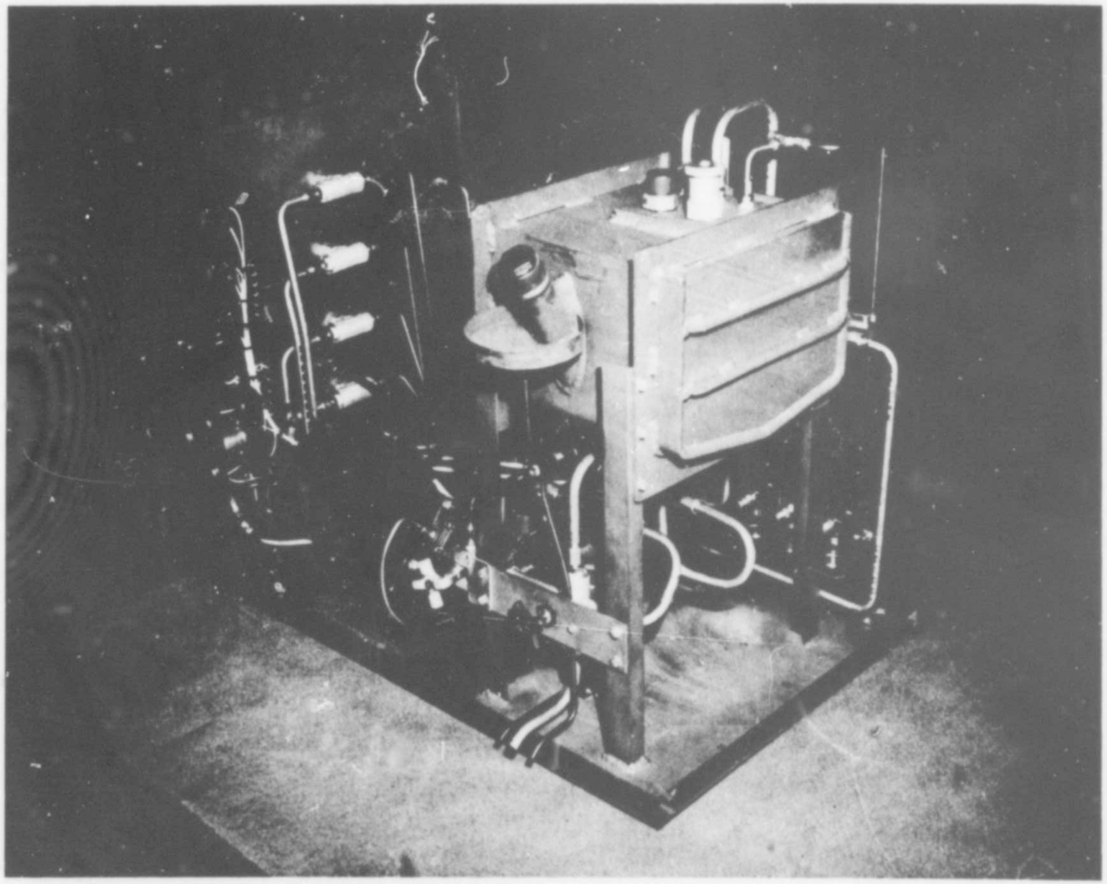


Figure 10. MSAS Power Supply Module.

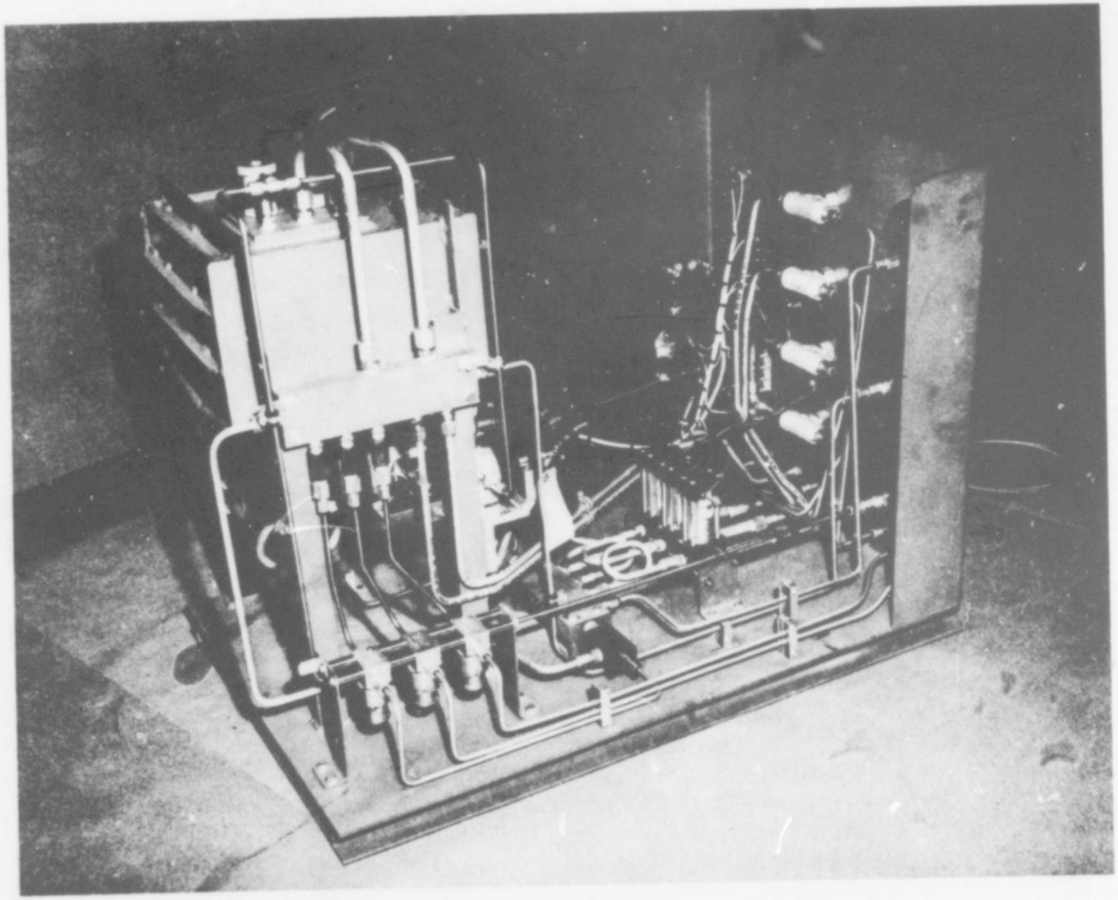


Figure 11. MSAS Power Supply Module.

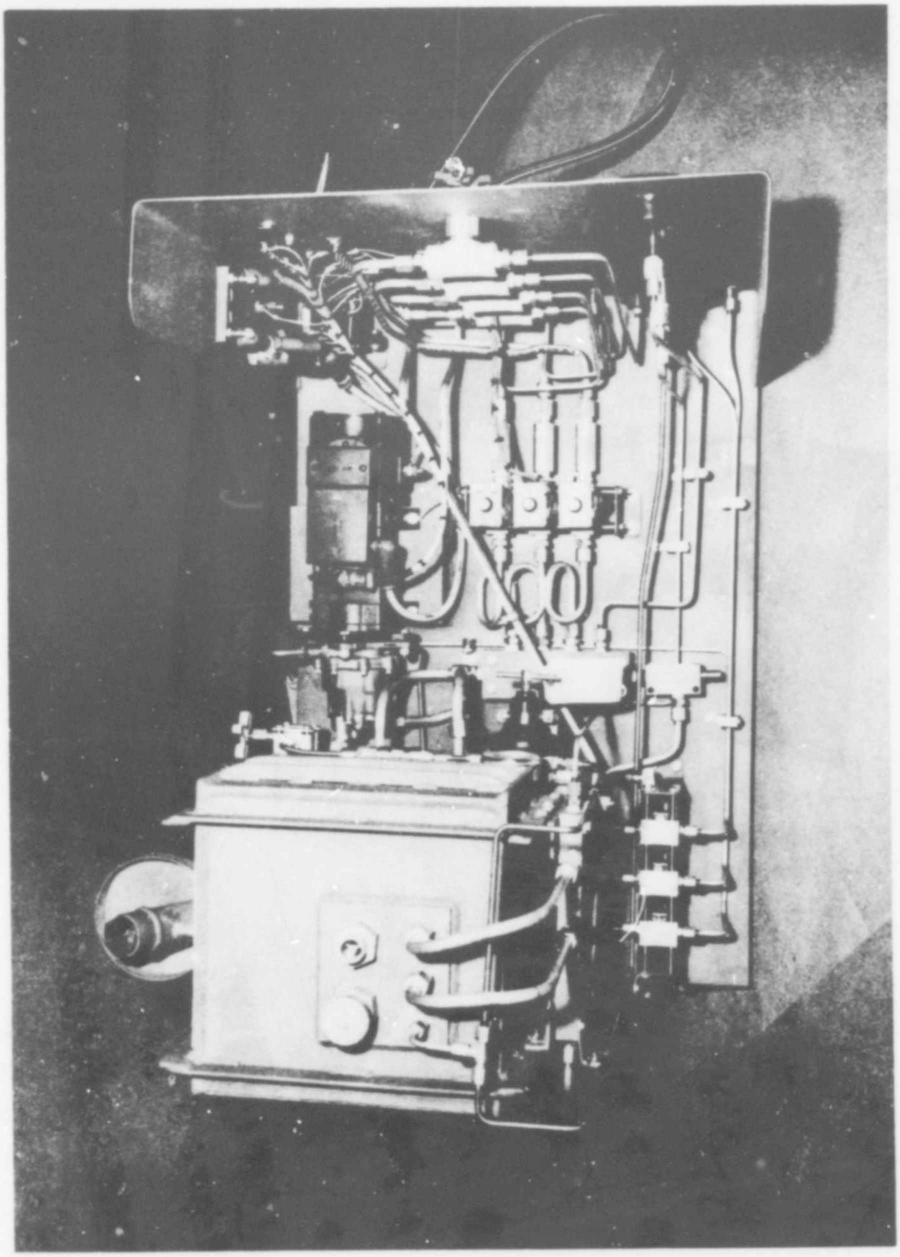


Figure 12. MSAS Power Supply Module.

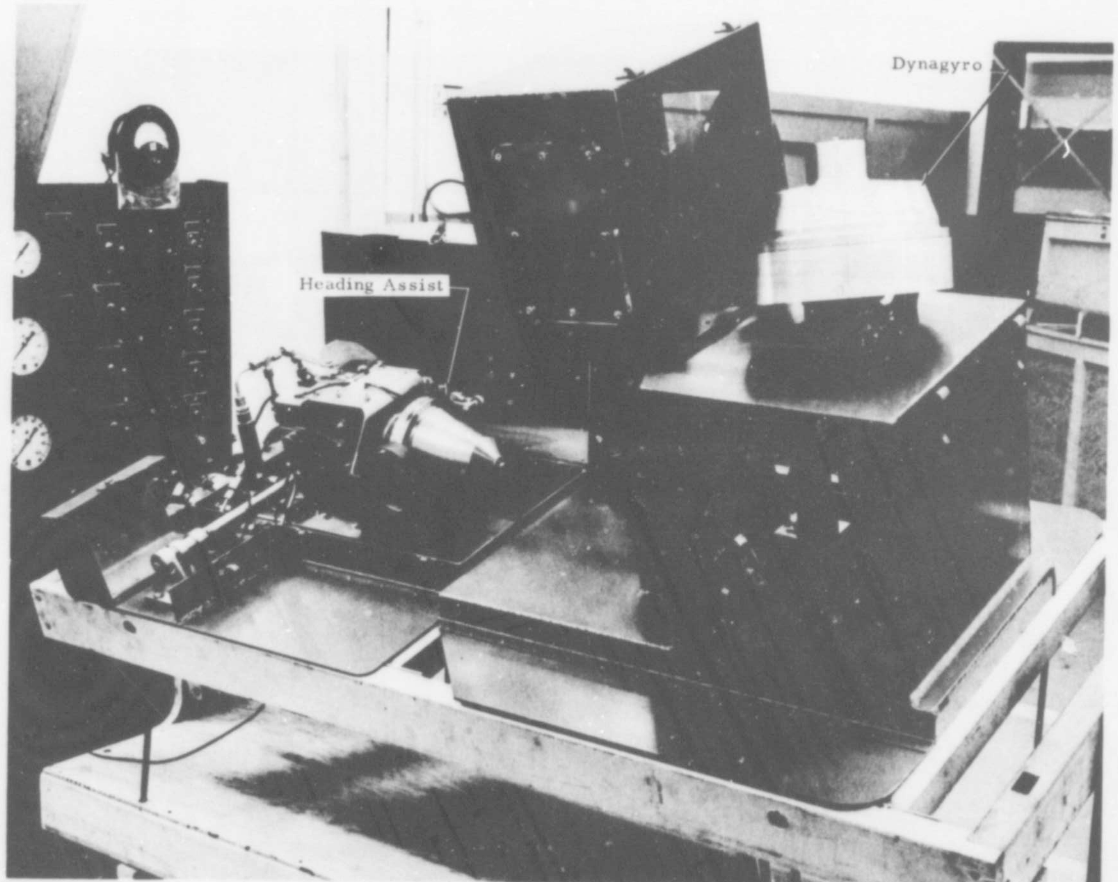


Figure 13. MSAS Modules.

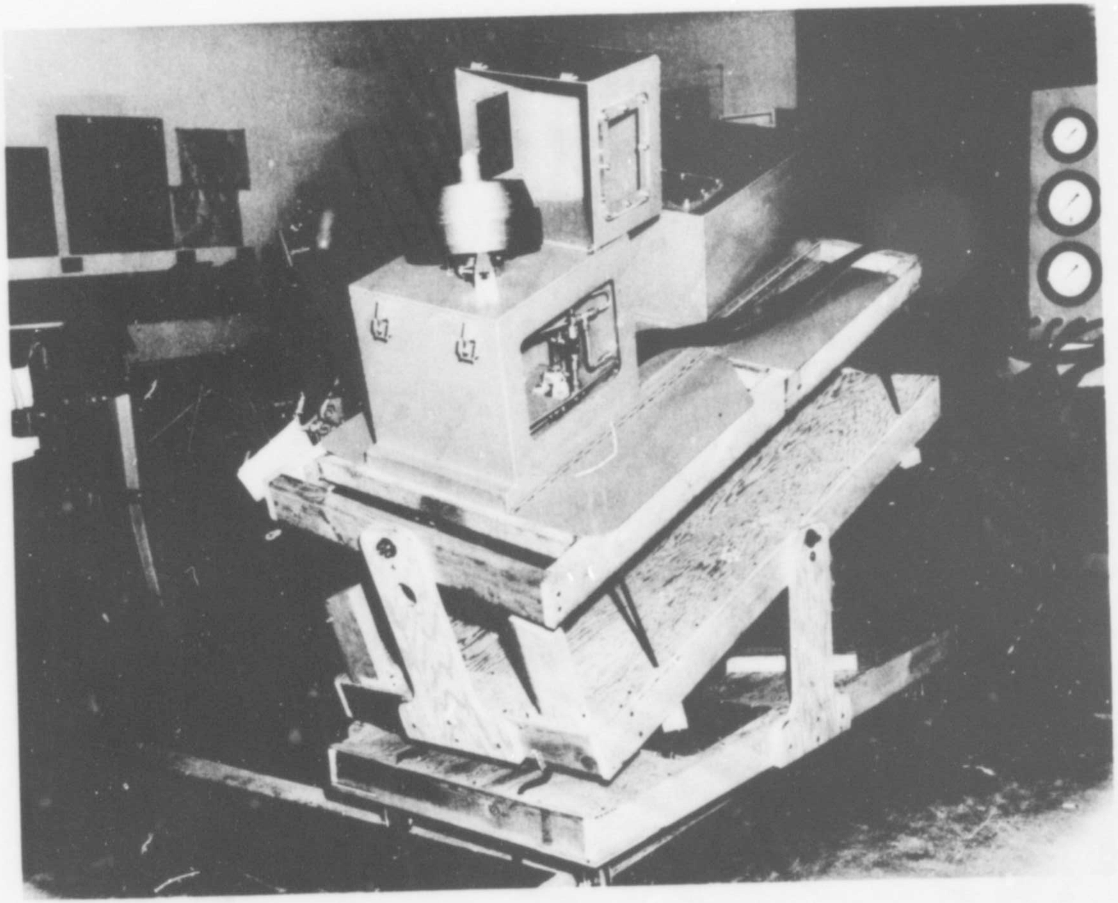


Figure 14. MSAS Modules on Three-Degree-of-Freedom Tilt Table.

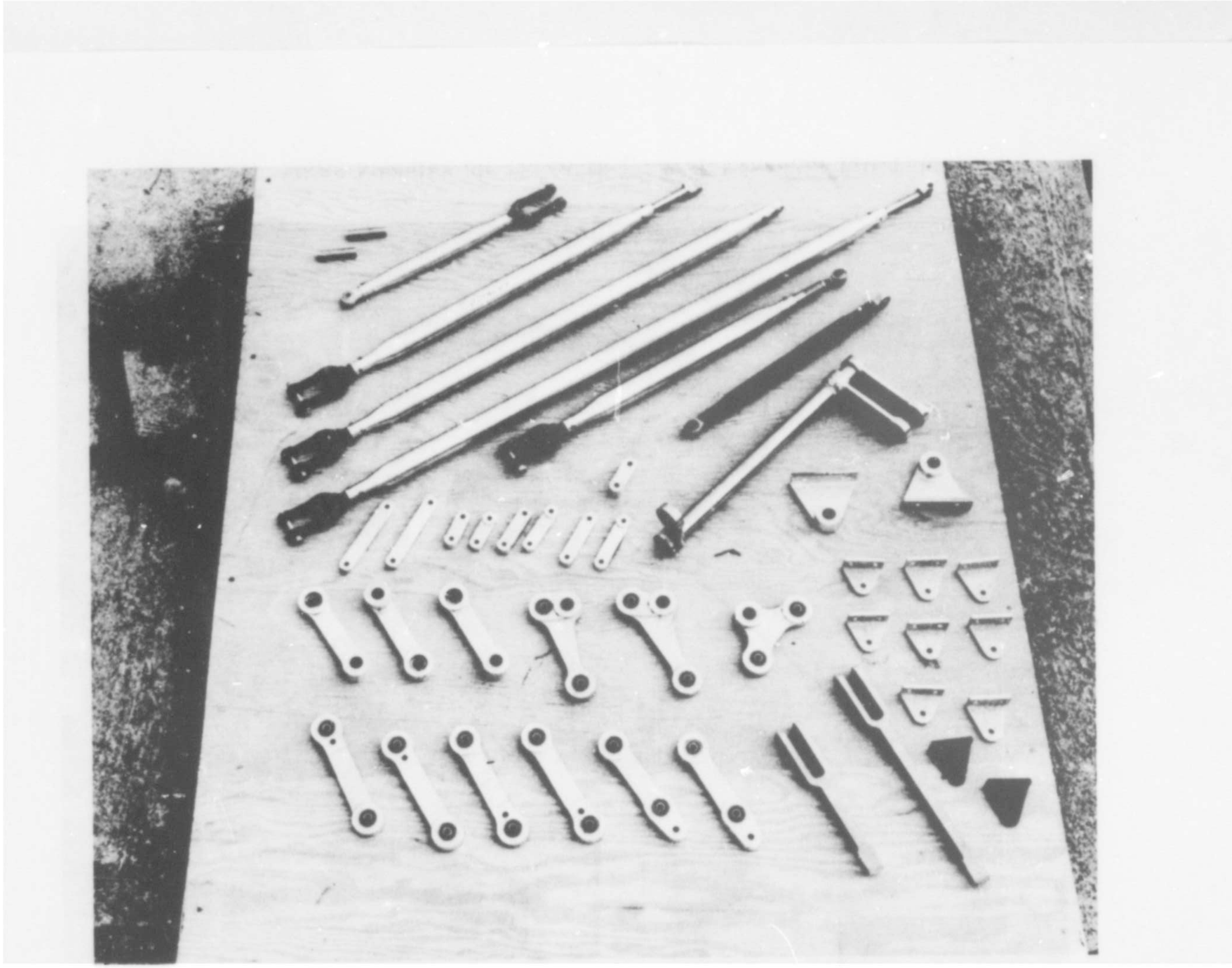


Figure 15. MSAS Components.

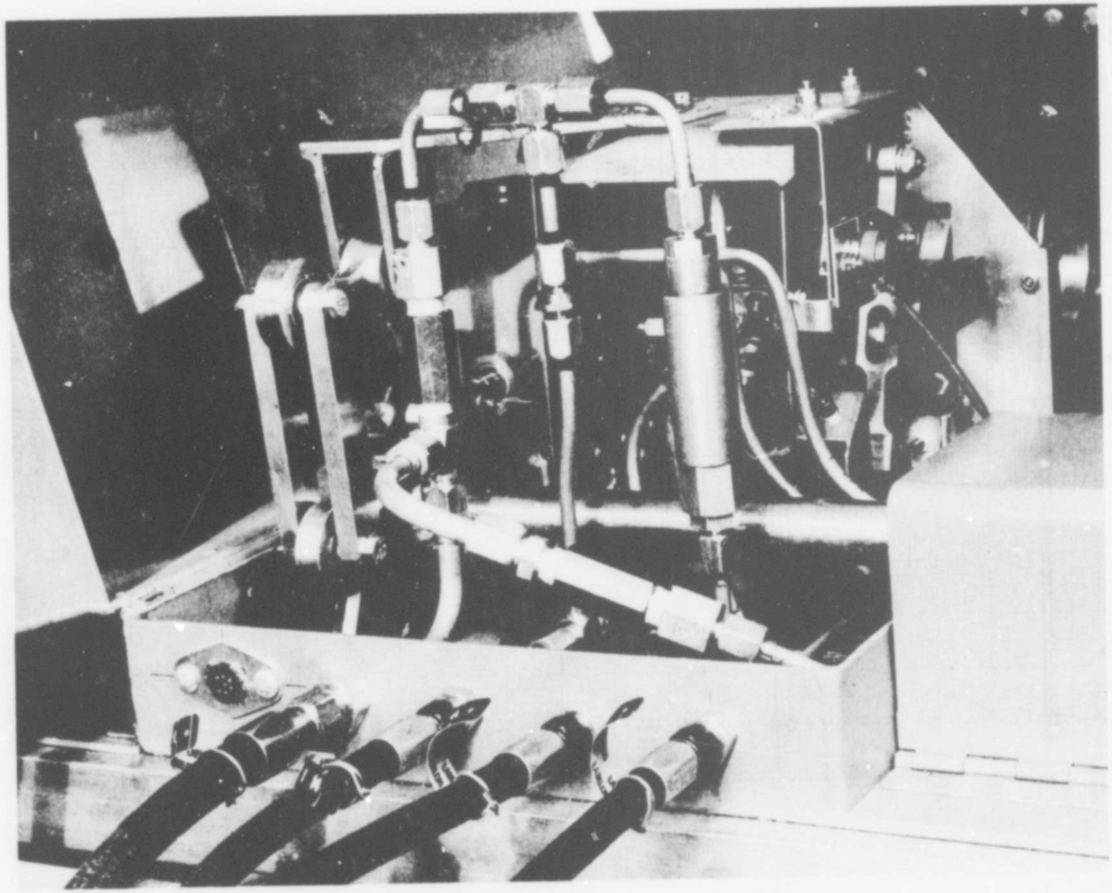
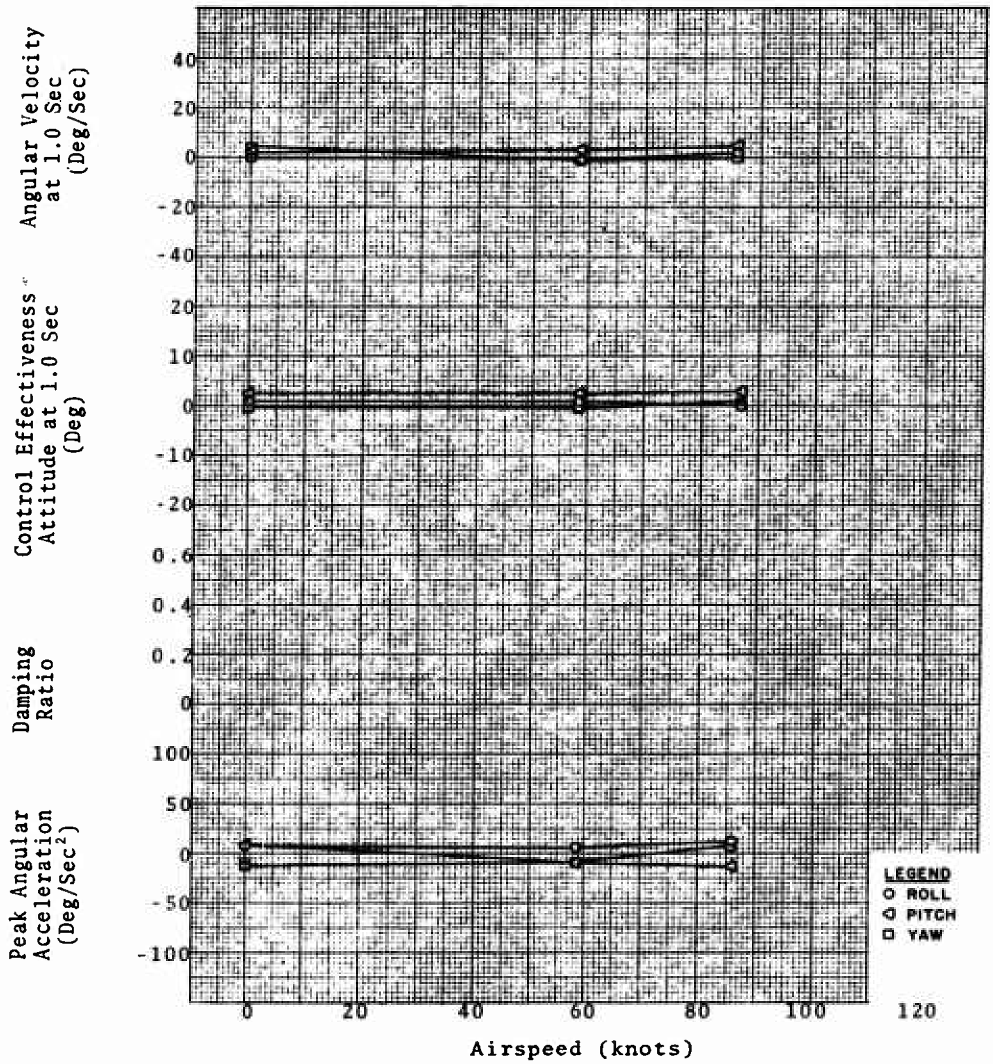


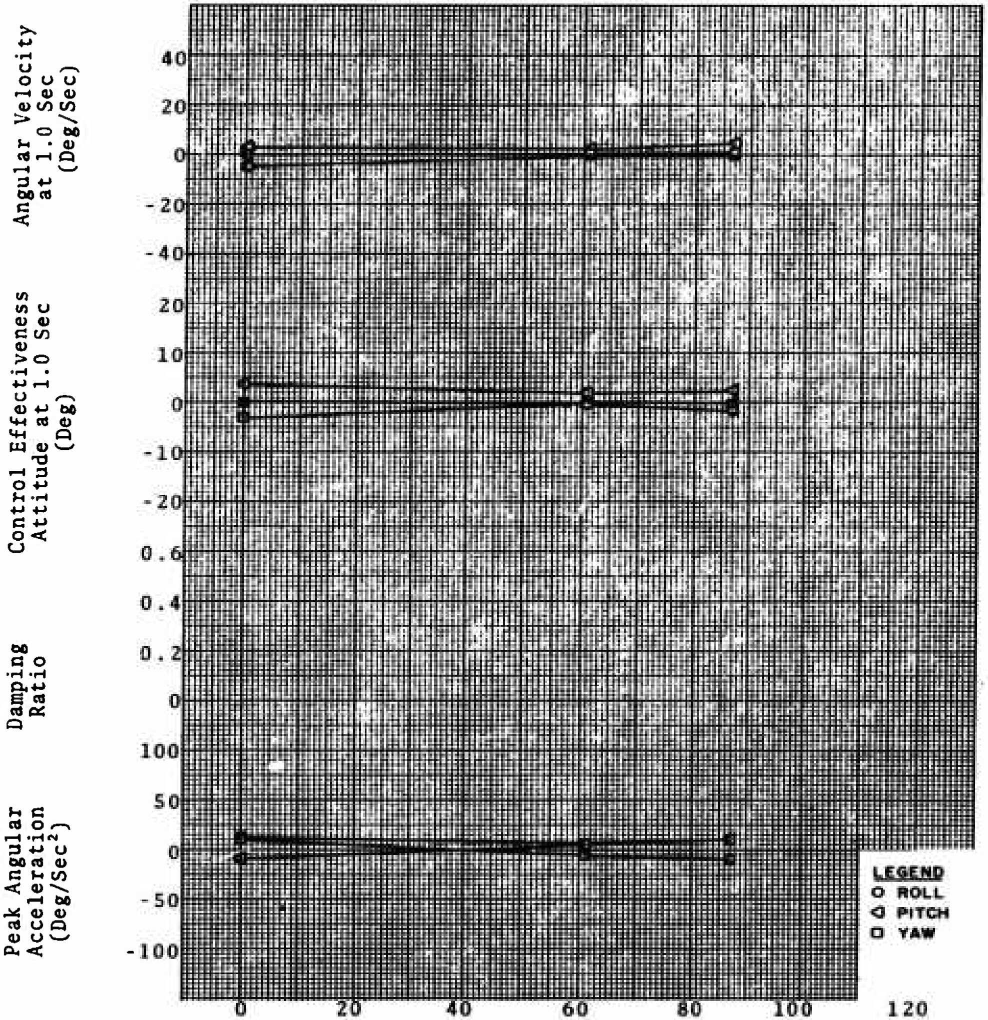
Figure 16. Yaw SAS Mechanism.



Configuration 1
Flt 6 Run 2,10,18

ESG.W. 7700 lb
ESC.G. 137
Rotor Speed 318 rpm

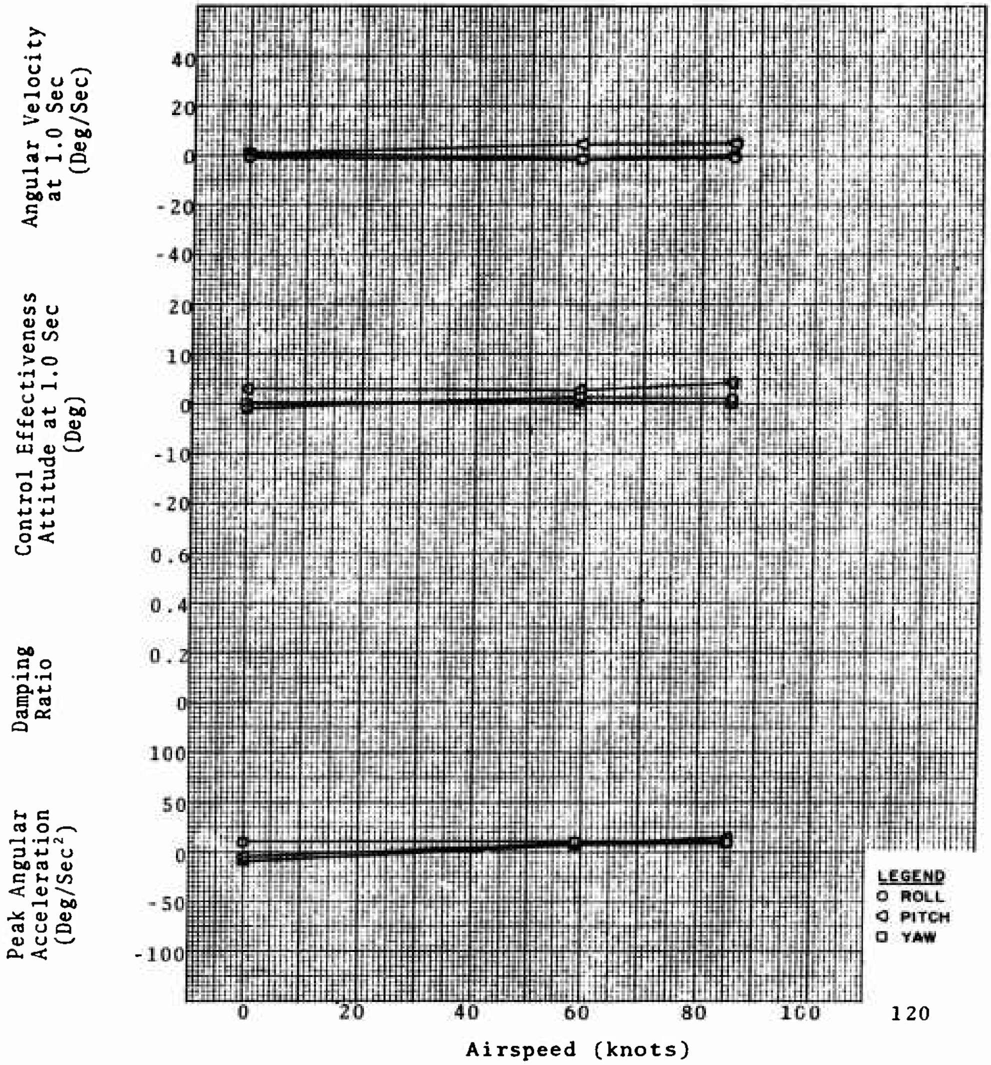
Figure 17. Aft Longitudinal Control Response for 1-Inch Pulse Input - MSAS Off (Configuration 1).



Configuration 3
Flt 7 Run 6,17
Flt 8 Run 3

ESG.W. 7700 lb
ESC.G. 137
Rotor Speed 318 rpm

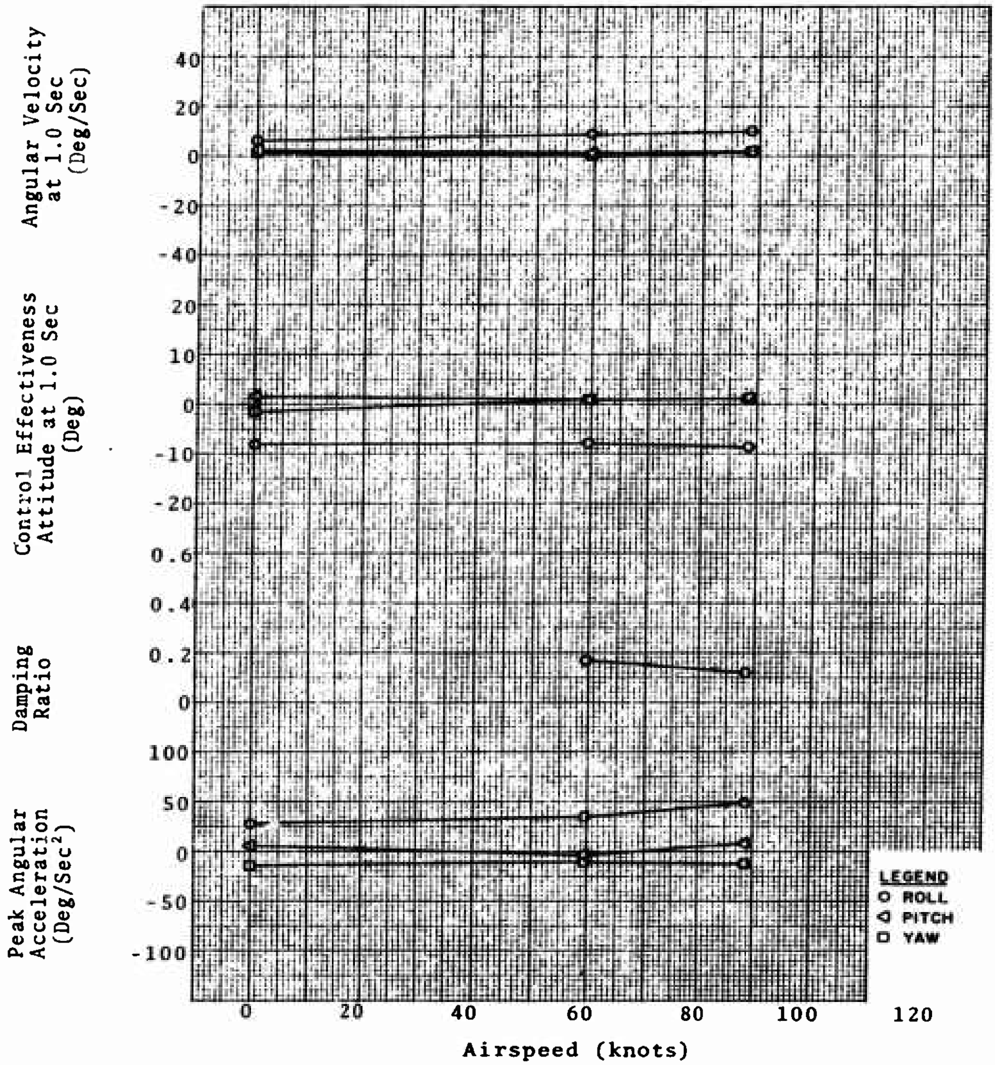
Figure 18. Aft Longitudinal Control Response for 1-Inch Pulse Input - MSAS Off (Configuration 3).



Configuration 5
Flt 7 Run 29,40
Flt 8 Run 20

ESG.W. 7700 lb
ESC.G. 137
Rotor Speed 318 rpm

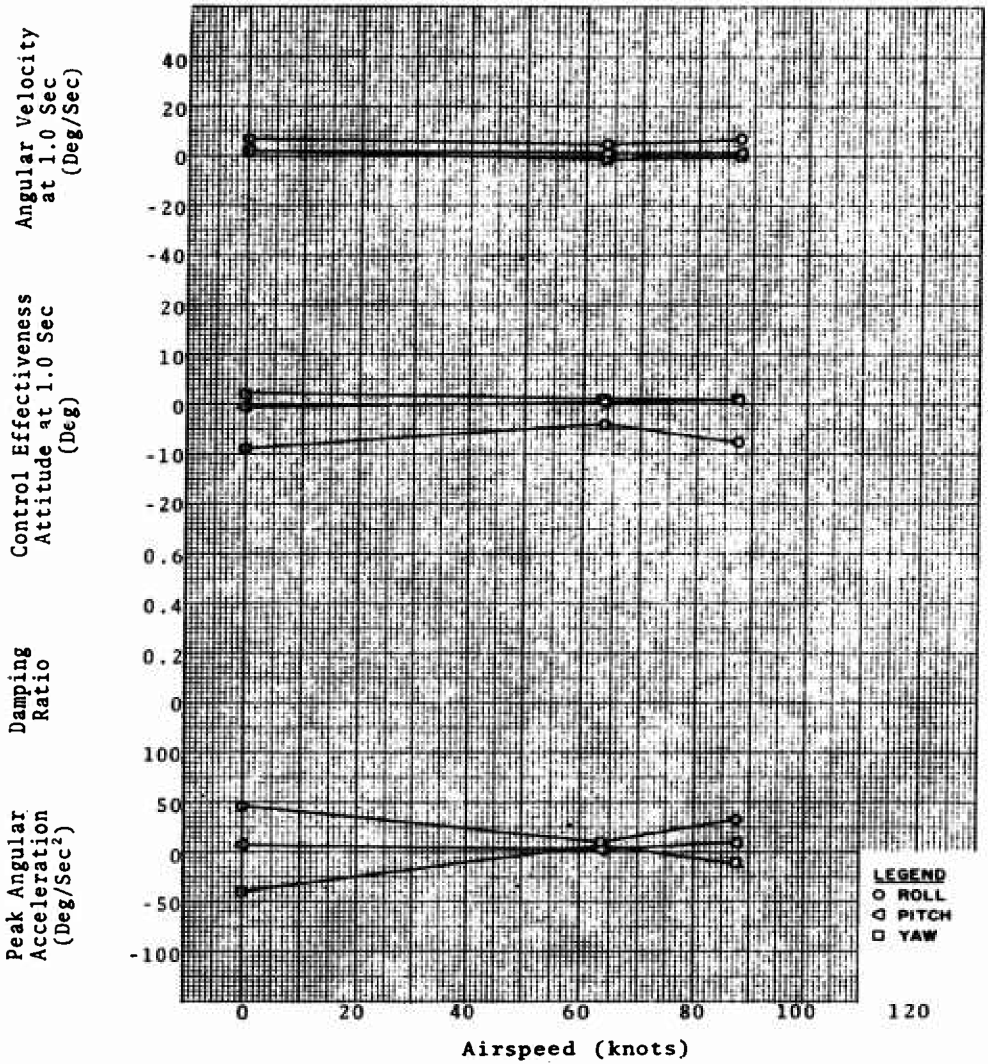
Figure 19. Aft Longitudinal Control Response for 1-Inch Pulse Input - MSAS On (Configuration 5).



Configuration 1
Flt 6 Run 4,12,20

ESG.W. 7700 lb
ESC.G. 137
Rotor Speed 318 rpm

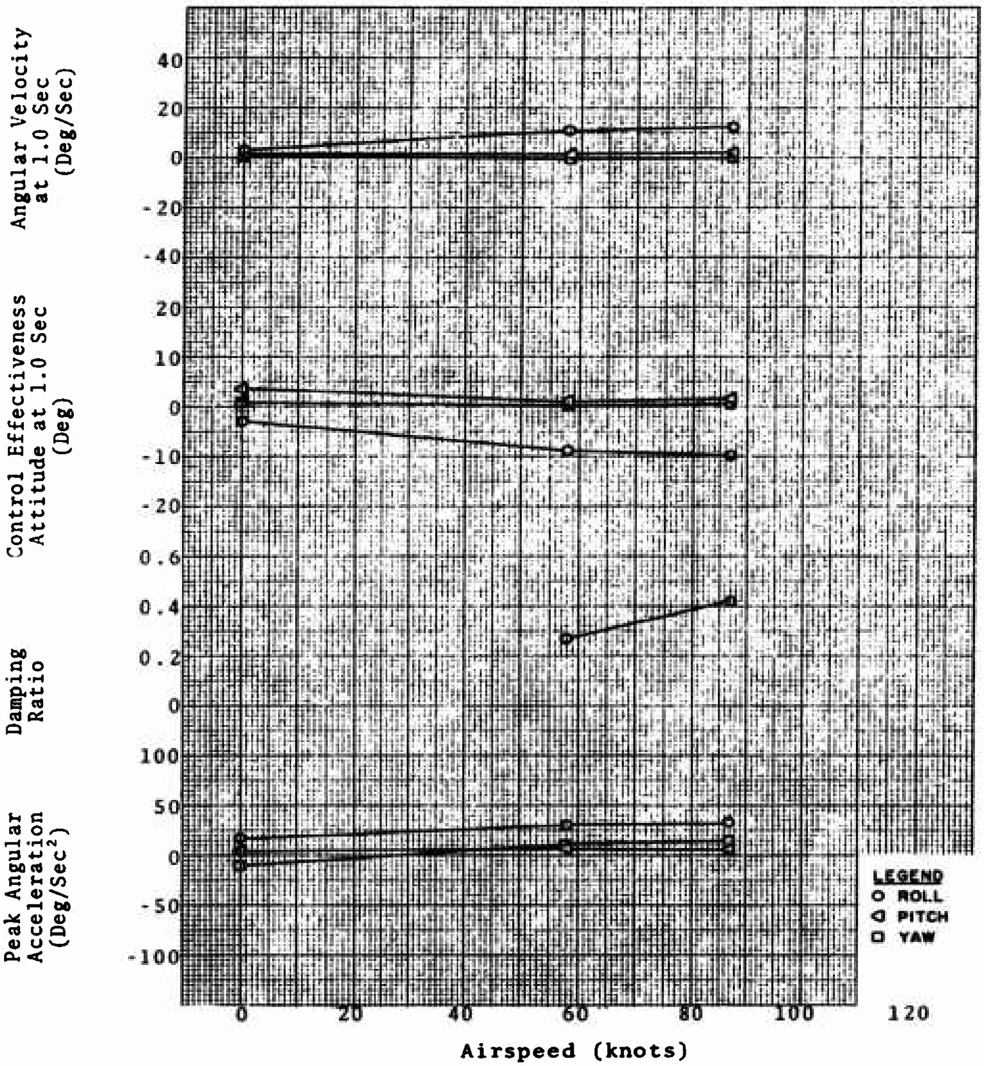
Figure 20. Left Lateral Control Response for 1-Inch Pulse Input - MSAS Off (Configuration 1).



Configuration 3
Flt 7 Run 7,20
Flt 8 Run 8

ESG.W. 7700 lb
ESC.G. 137
Rotor Speed 318 rpm

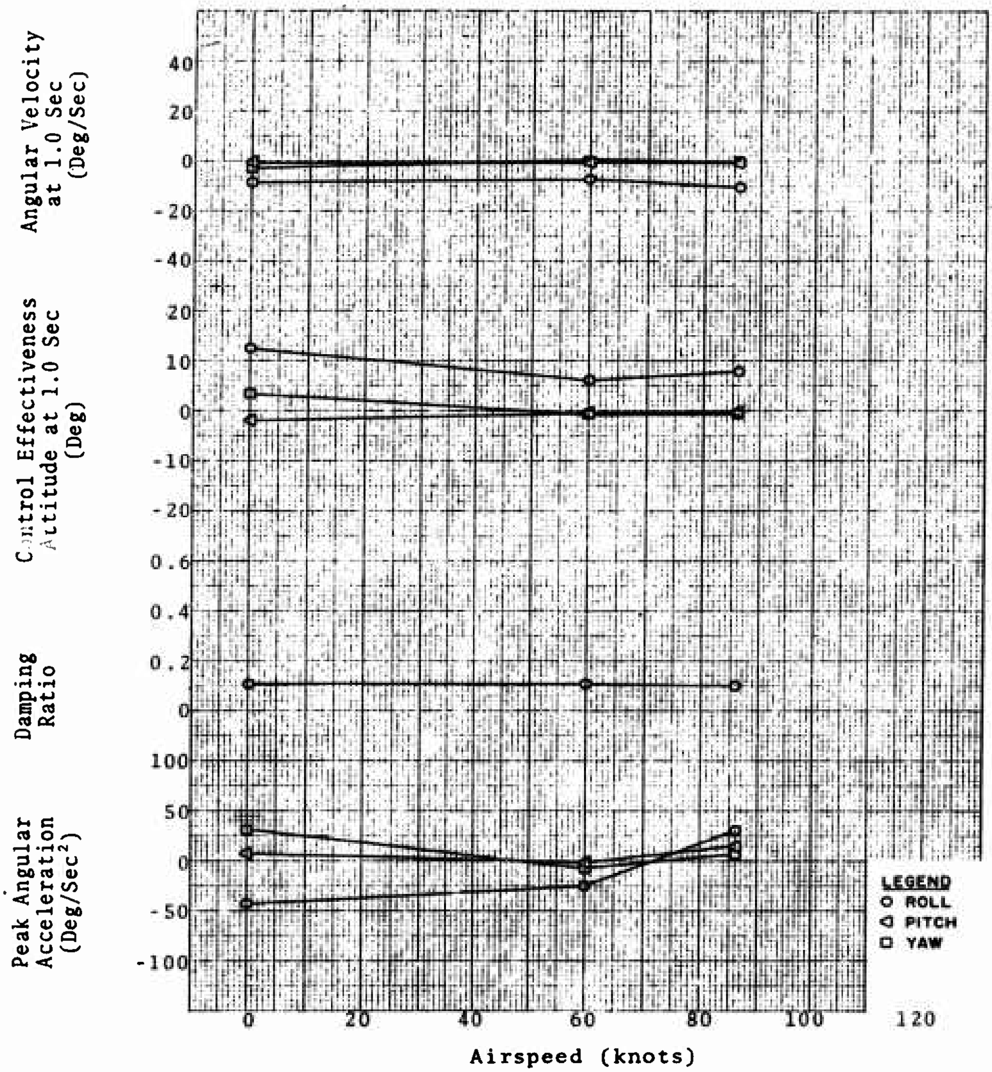
Figure 21. Left Lateral Control Response for 1-Inch Pulse Input - MSAS Off (Configuration 3).



Configuration 5
Flt 7 Run 31,42
Flt 8 Run 24

ESG.W. 7700 lb
ESC.G. 137
Rotor Speed 318 rpm

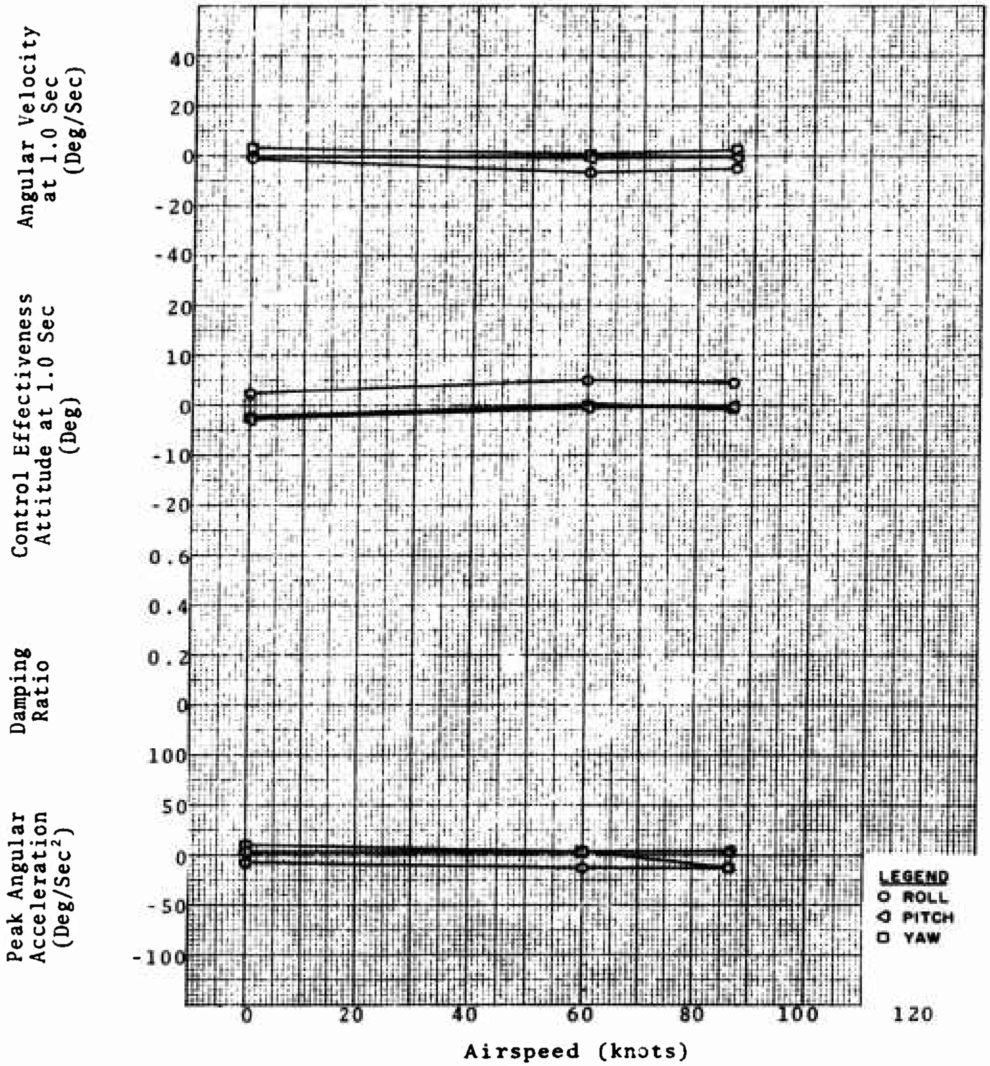
Figure 22. Left Lateral Control Response for 1-Inch Pulse Input - MSAS On (Configuration 5).



Configuration 1 | Flt 6 | Run 5,13,22

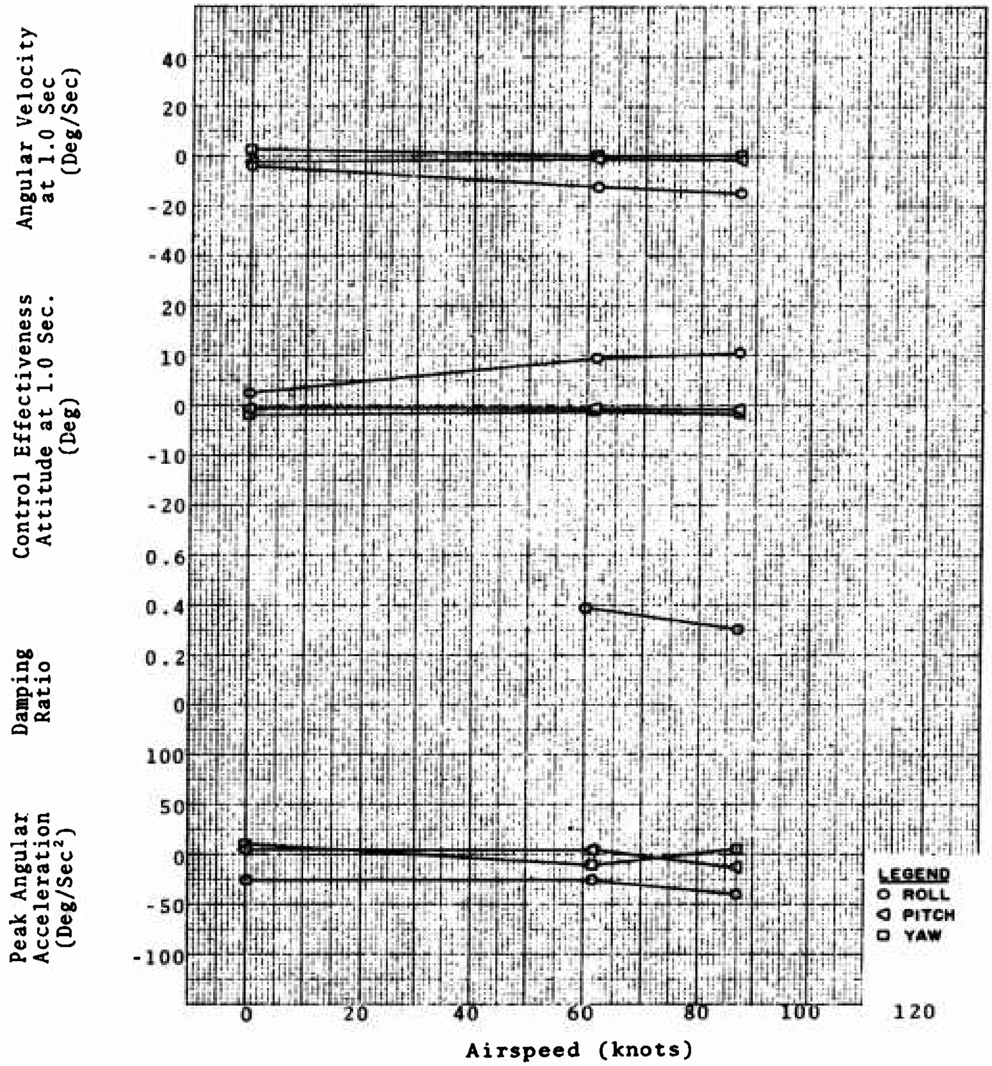
ESG.W. 7700 lb
ESC.G. 137
Rotor Speed 318 rpm

Figure 23. Right Lateral Control Response for 1-Inch Pulse Input - MSAS Off (Configuration 1).



Configuration 3		ESG.W.	7700 lb
Flt 7	Run 8	ESC.G.	137
Flt 8	Run 10	Rotor Speed	318 rpm

Figure 24. Right Lateral Control Response for 1-Inch Pulse Input - MSAS Off (Configuration 3).



Configuration 5
Flt 7 Run 32,43
Flt 8 Run 26

ESG.W. 7700 lb
ESC.G. 137
Rotor Speed 318 rpm

Figure 25. Right Lateral Control Response for 1-Inch Pulse Input - MSAS On (Configuration 5).

LITERATURE CITED

1. George, M., et al., DYNAGYRO - A MECHANICAL STABILITY AUGMENTATION SYSTEM FOR HELICOPTERS; Dynasciences Corp., USAAVLABS Technical Report 67-10, U.S. Army Aviation Materiel Laboratories, Fort Eustis, Virginia, March 1967, AD 654046.

2. George, M., et al., RELIABILITY EVALUATION OF A MECHANICAL STABILITY AUGMENTATION SYSTEM FOR HELICOPTERS, Dynasciences Corp., USAAVLABS Technical Report 69-17, U.S. Army Aviation Materiel Laboratories, Fort Eustis, Virginia, June 1969, AD 855874.

APPENDIX I

TEST DATA

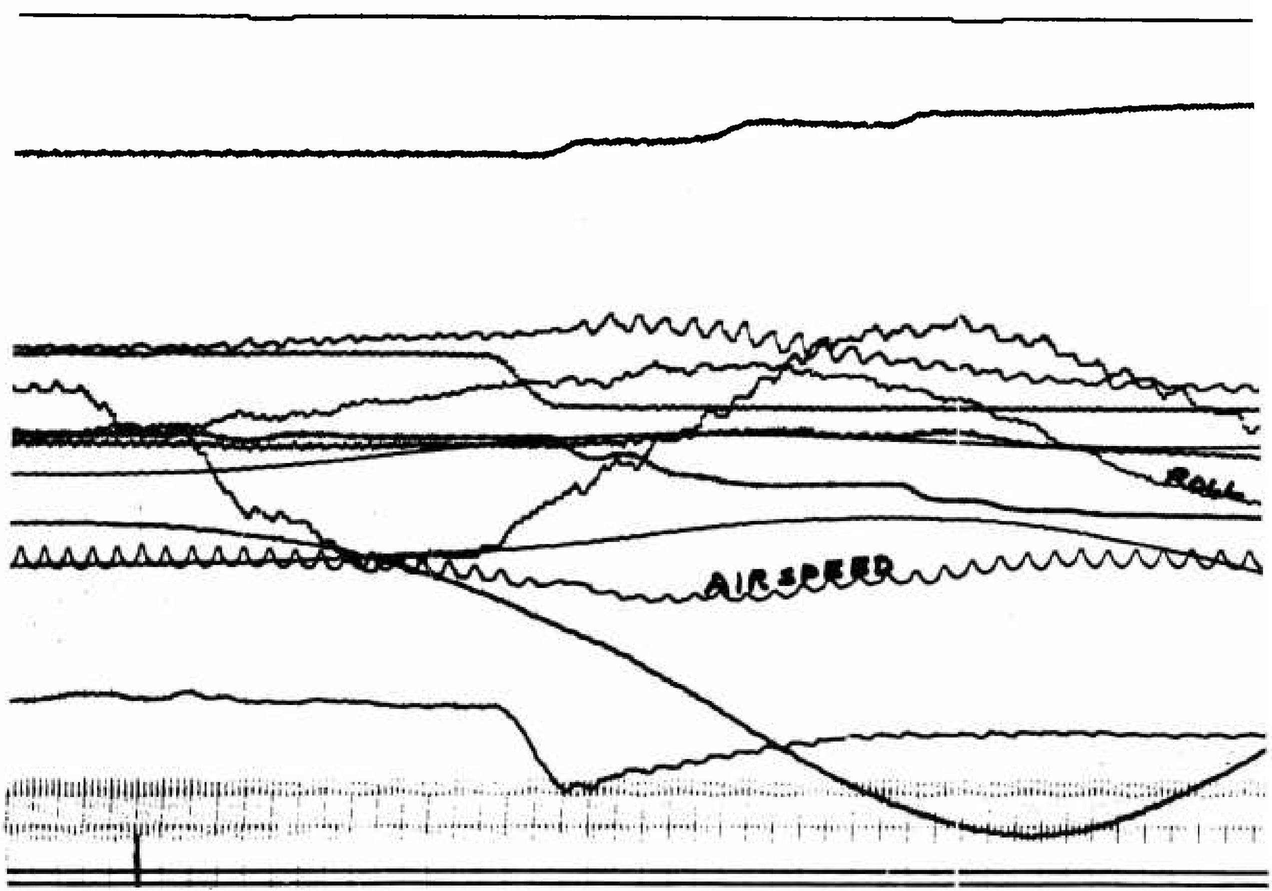
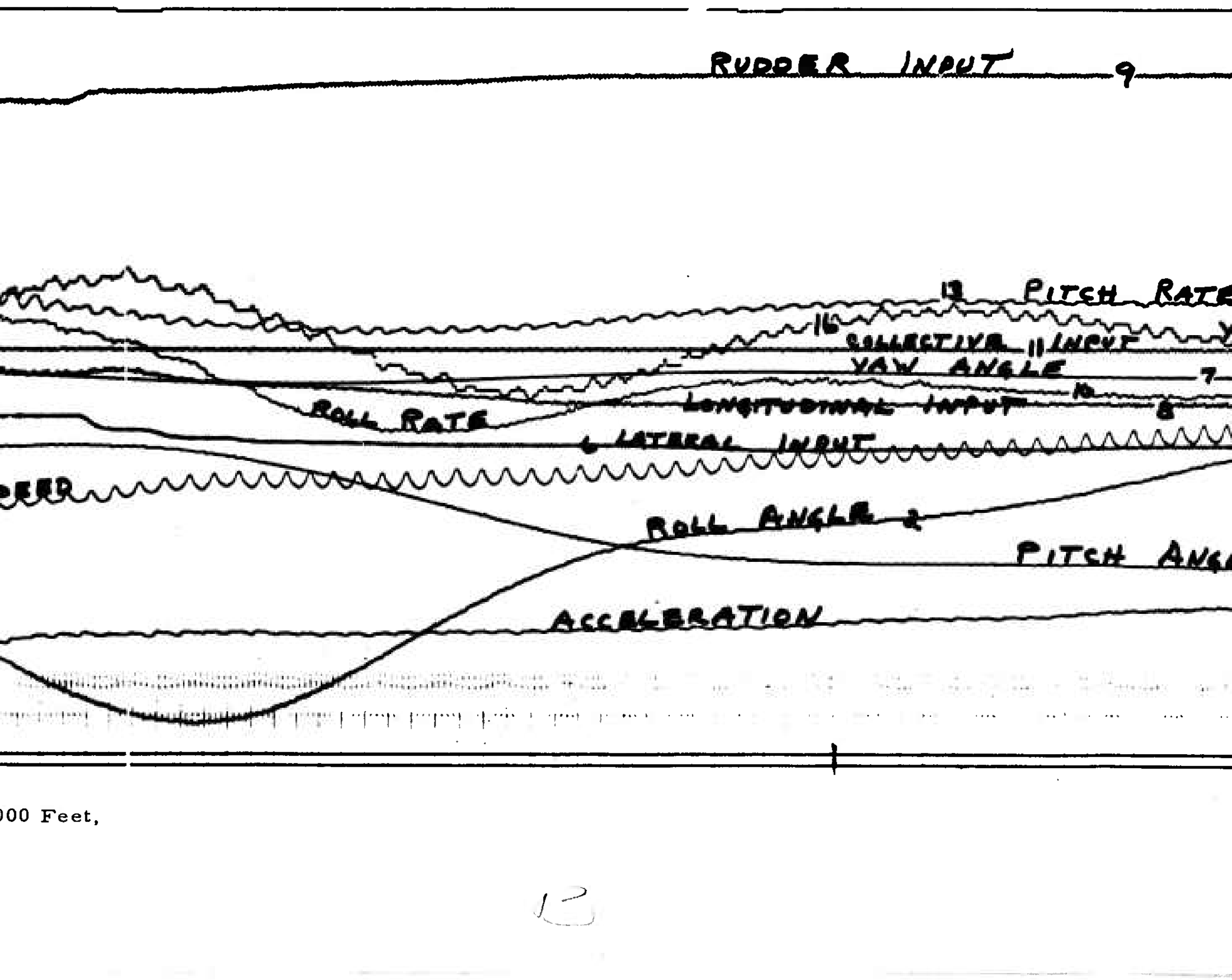
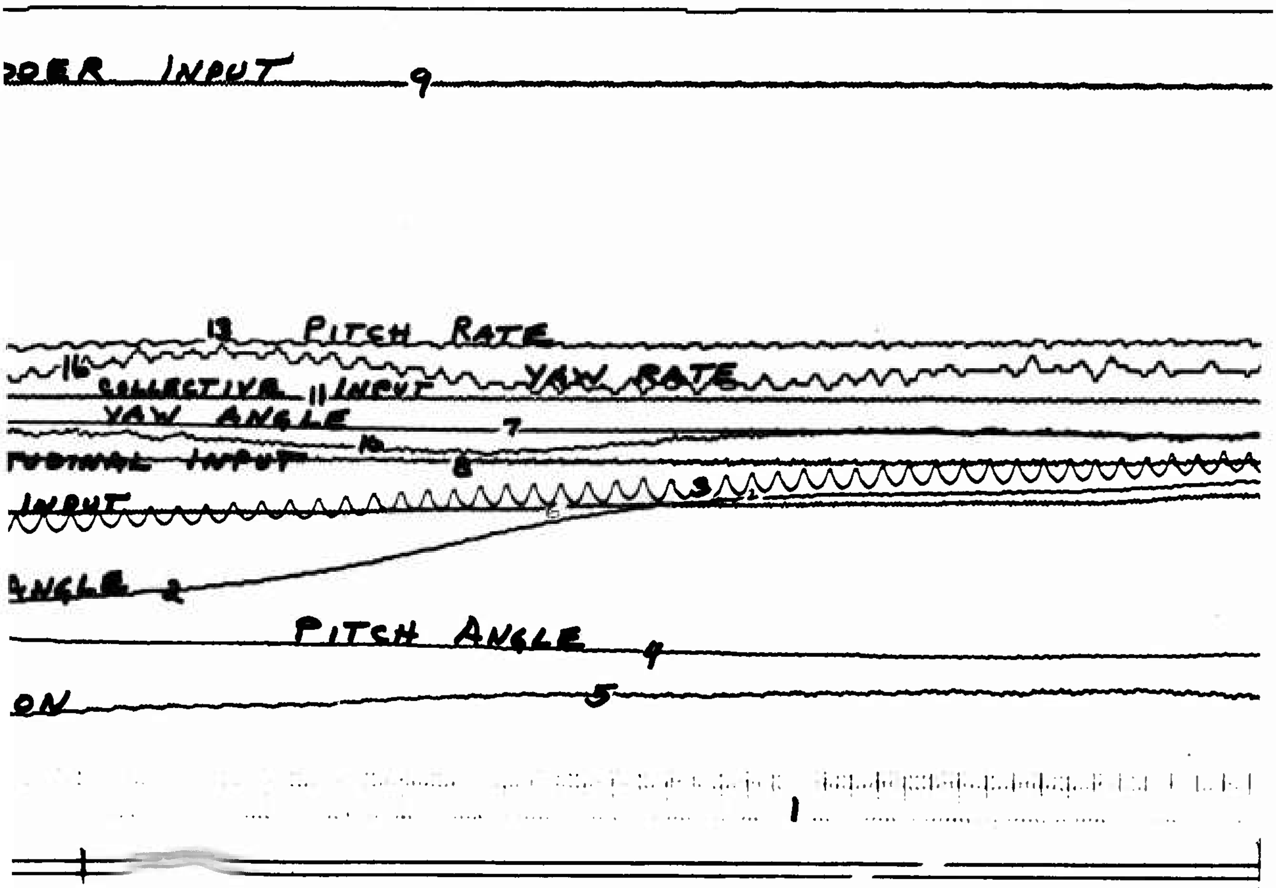


Figure 26. Autorotational Entry - 60 KIAS, 4000 Feet, With Stabilizer Bar.

A


RUDDER INPUT
9
13
PITCH
16
COLLECTIVE
11 INPUT
YAW ANGLE
7
10
LONGITUDINAL INPUT
8
ROLL RATE
6 LATERAL INPUT
ROLL ANGLE 2
ACCELERATION

000 Feet,



INPUT
9
13
PITCH RATE
16
YAW RATE
11
INPUT
YAW ANGLE
7
10
8
INPUT
3
2
PITCH ANGLE
4
5
1

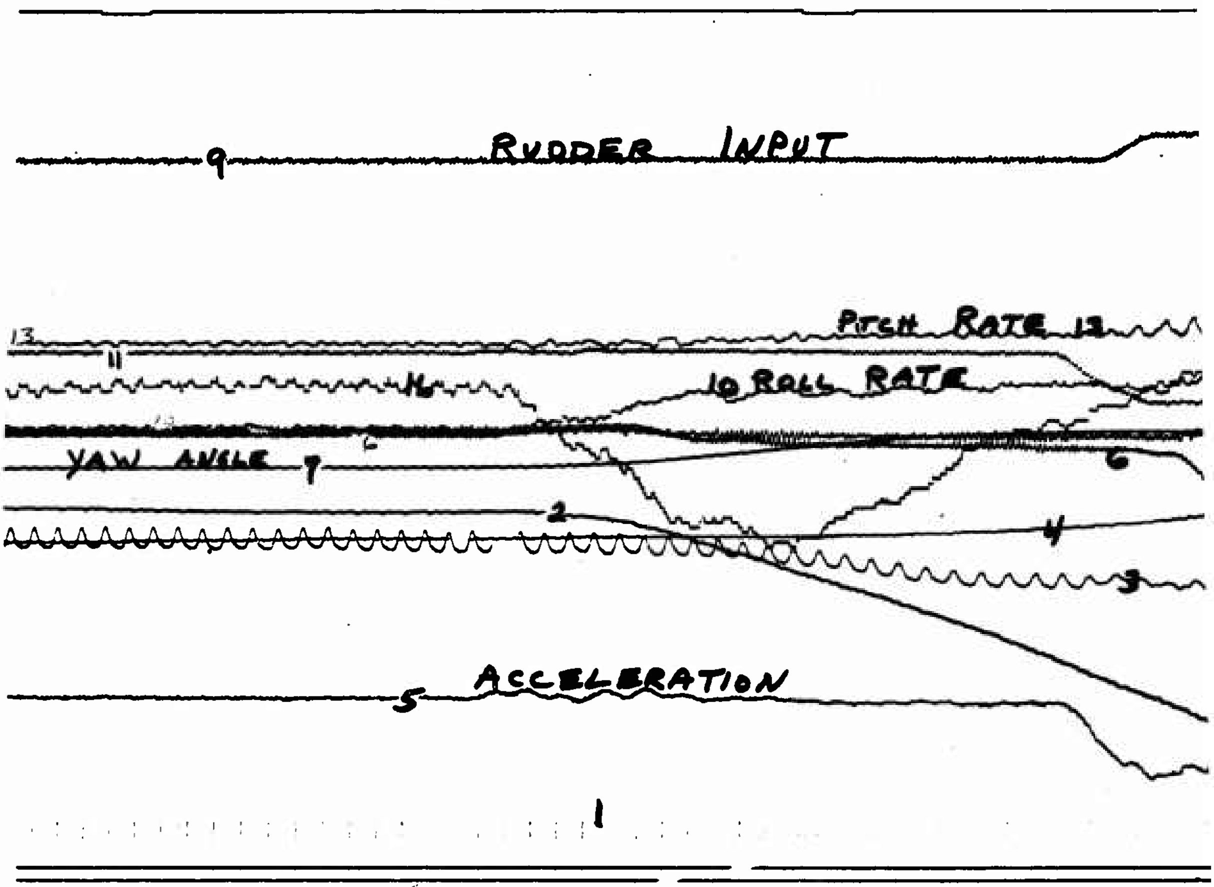
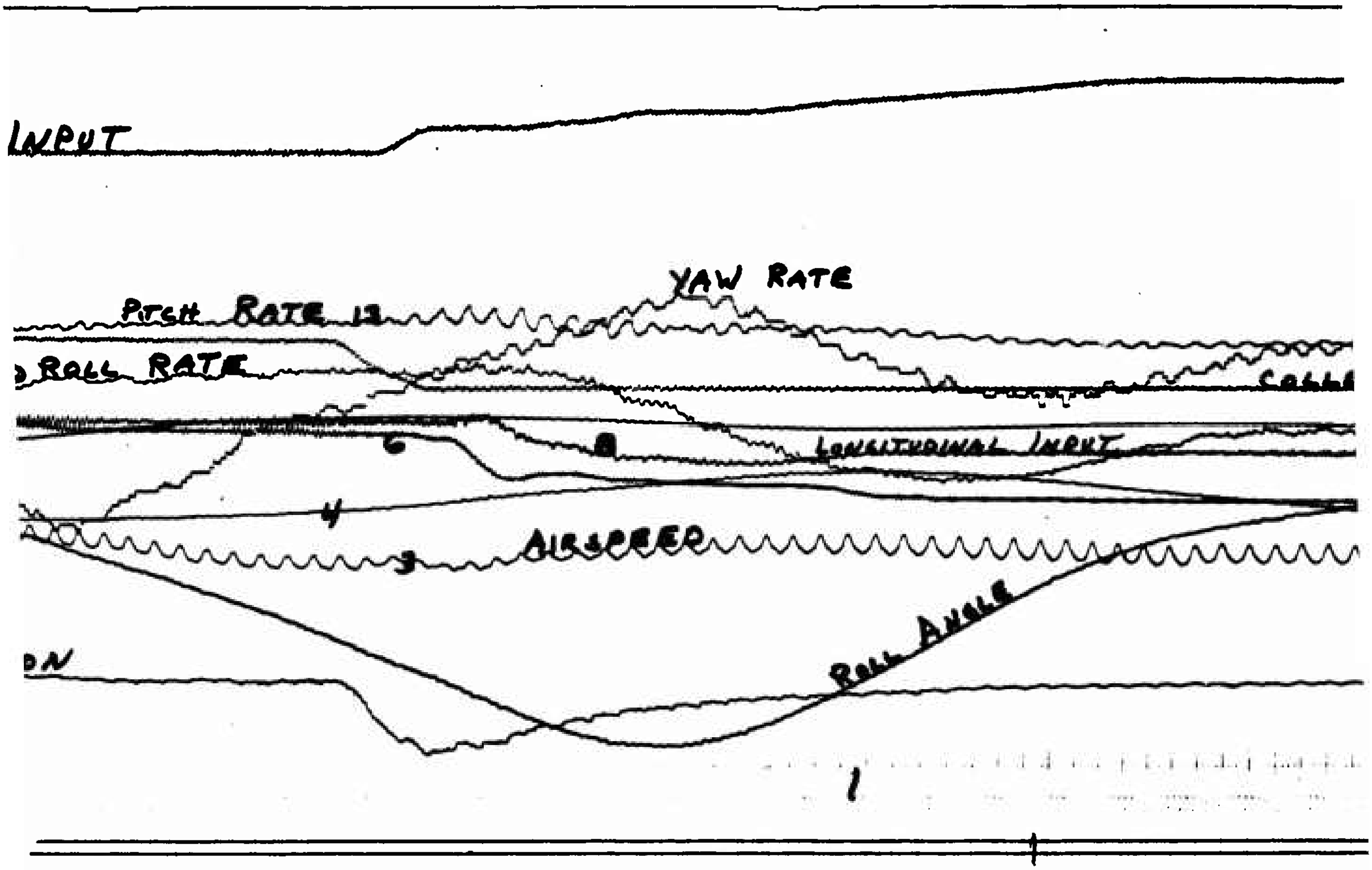


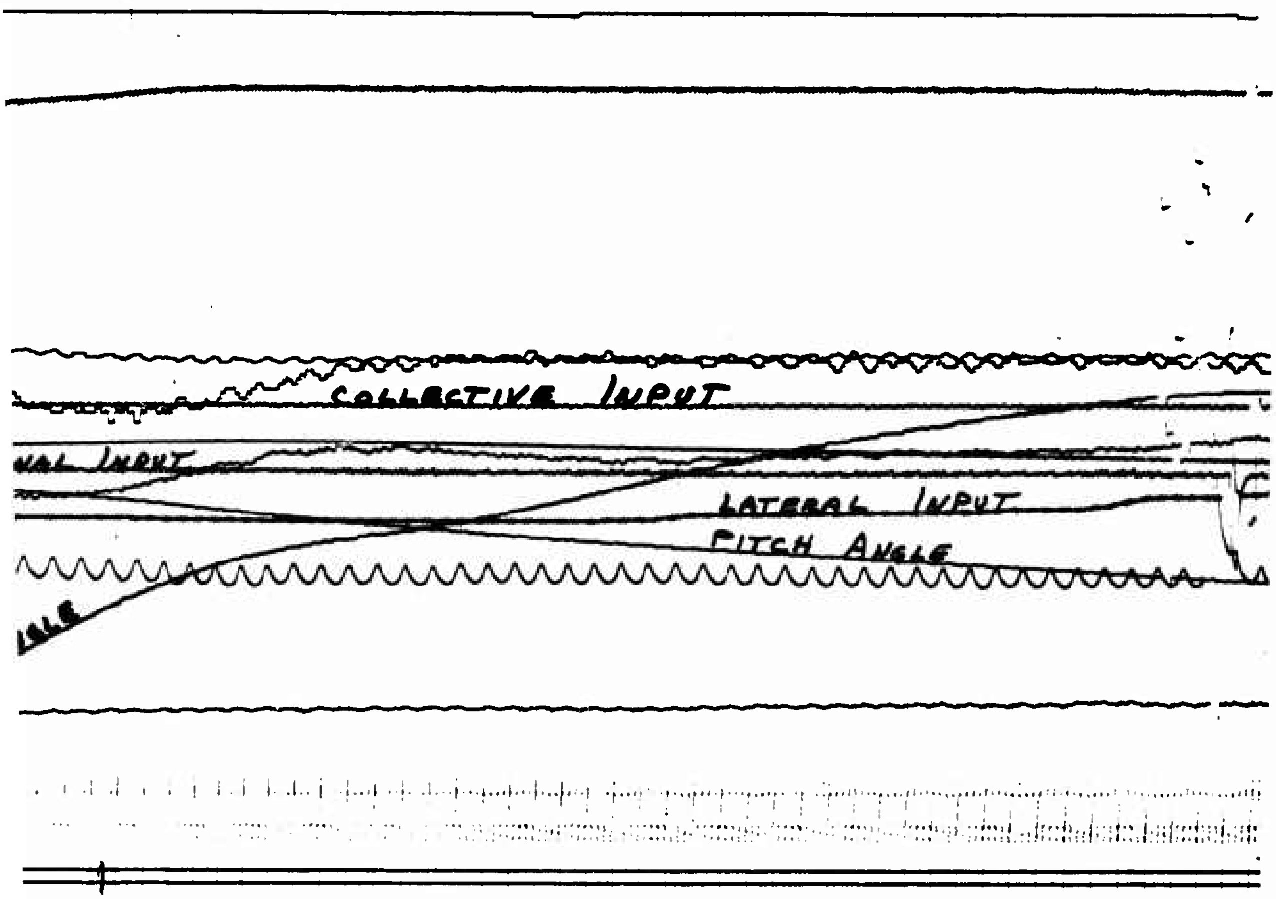
Figure 27. Autorotational Entry - 60 KIAS, 4000 Feet, With Yaw SAS.

A


INPUT
YAW RATE
PITCH RATE 12
ROLL RATE
6
8
LONGITUDINAL INPUT
4
AIRSPEED
3
ROLL ANGLE
ON
1

4000 Feet,

B



C

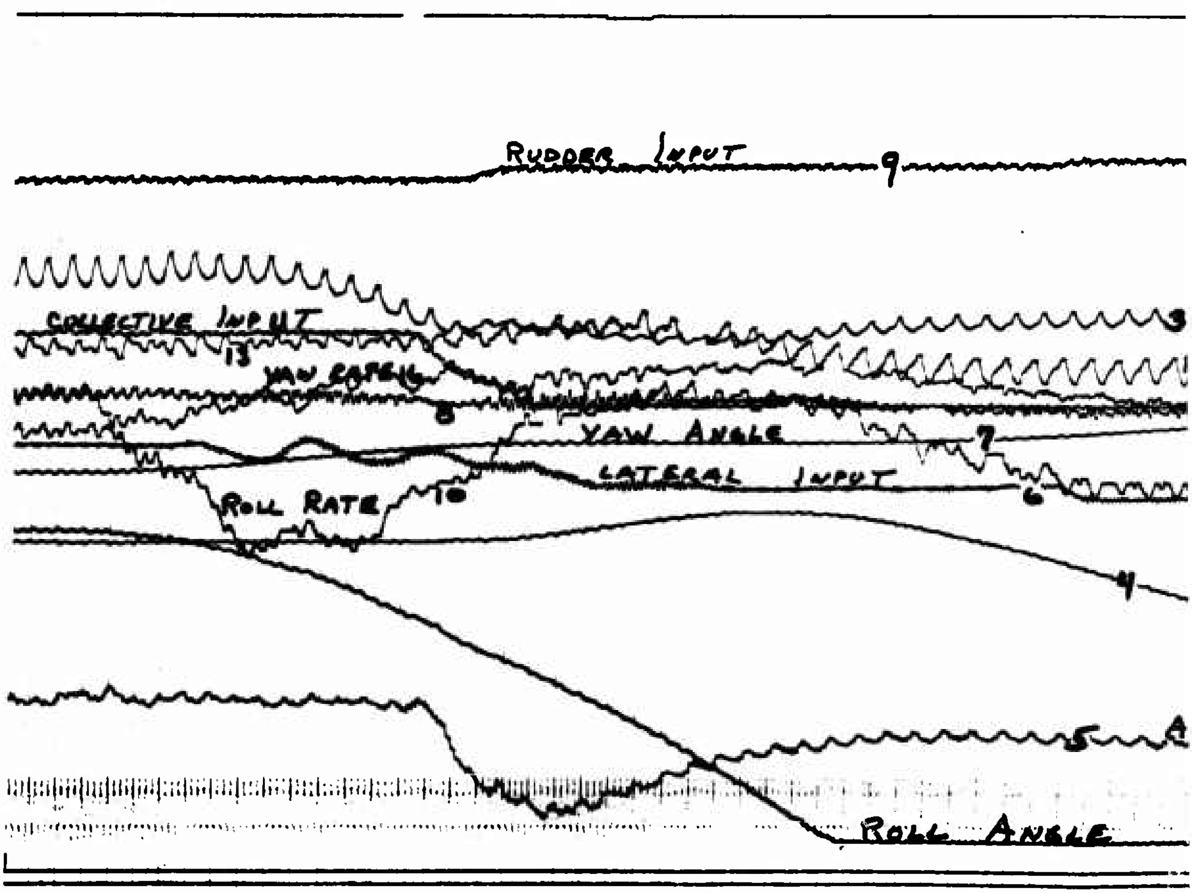
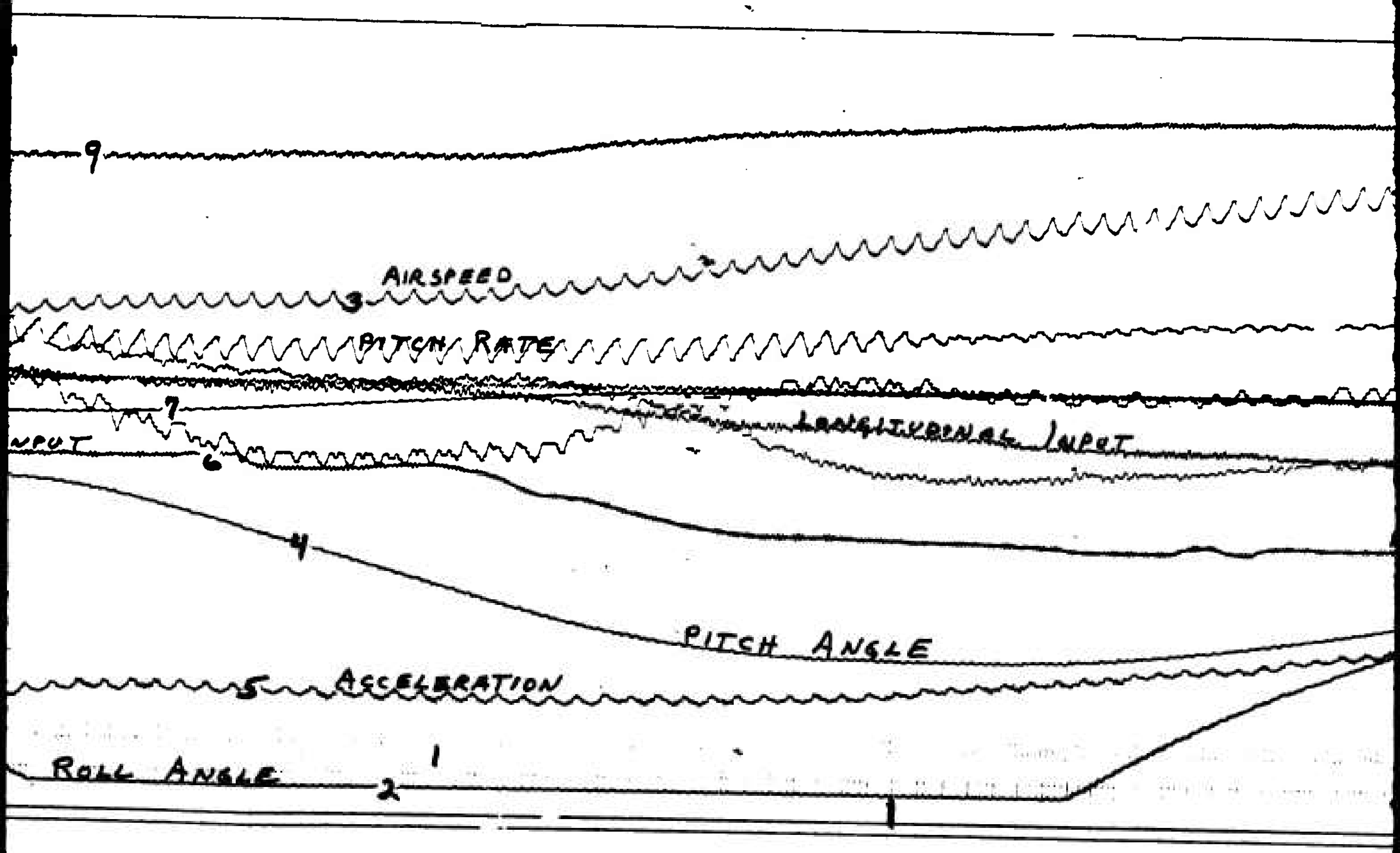
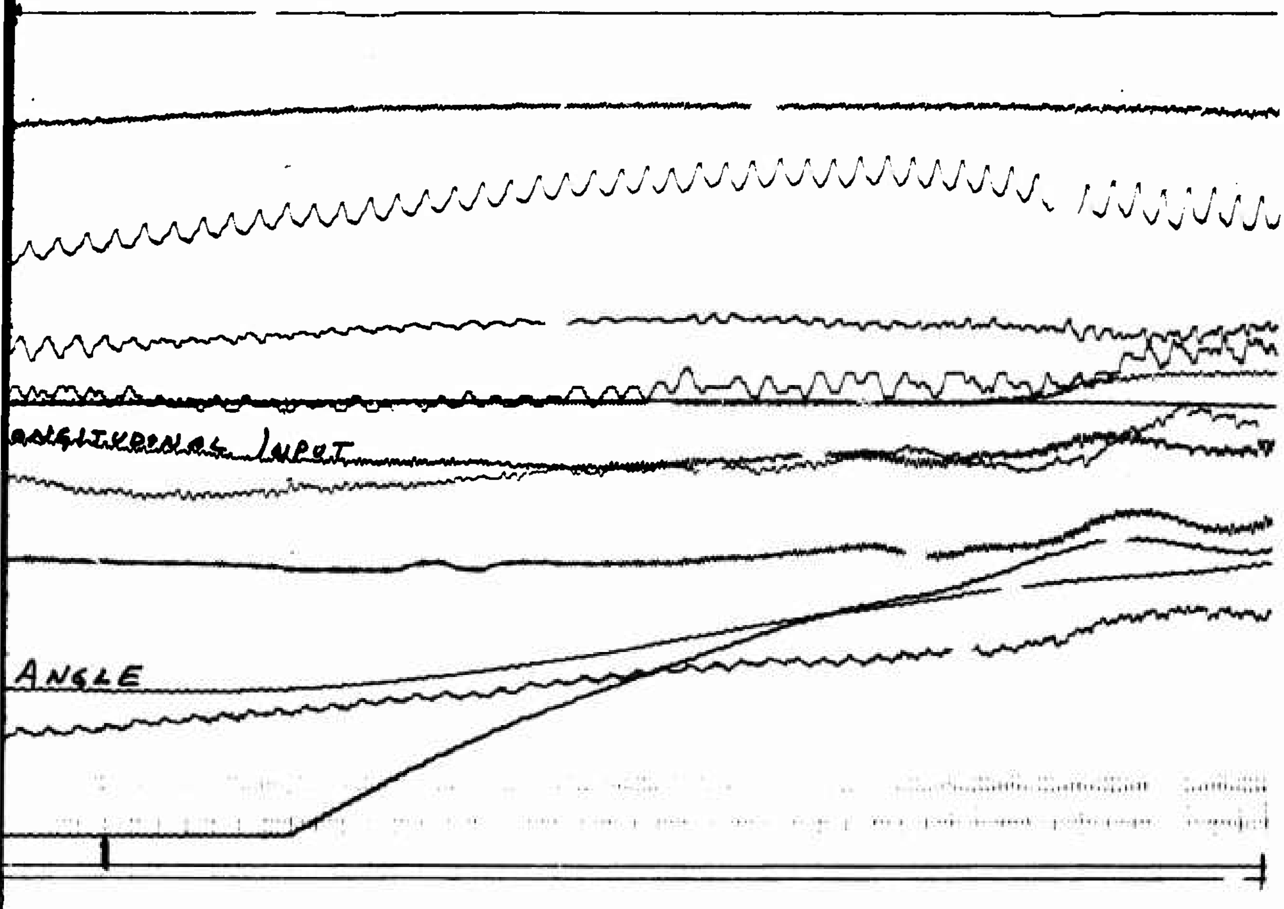


Figure 28. Autorotational Entry - 90 KIAS, 4000 Feet, With Stabilizer Bar.

A


9
AIRSPEED
3
PITCH RATE
7
LONGITUDINAL INPUT
NPUT
6
4
PITCH ANGLE
5
ACCELERATION
1
ROLL ANGLE
2
1

eet,



ONGITUDINAL INPUT
ANGLE

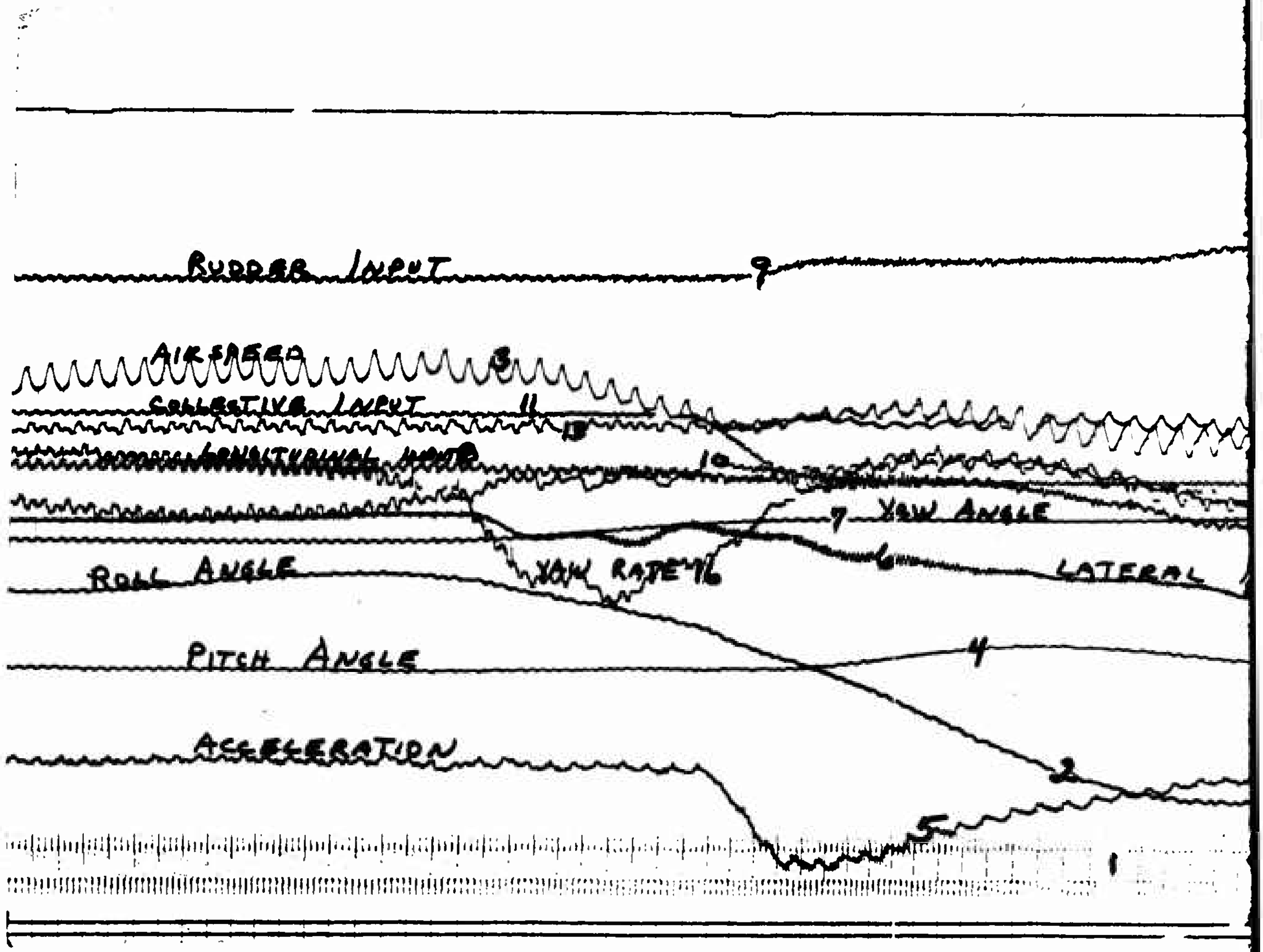
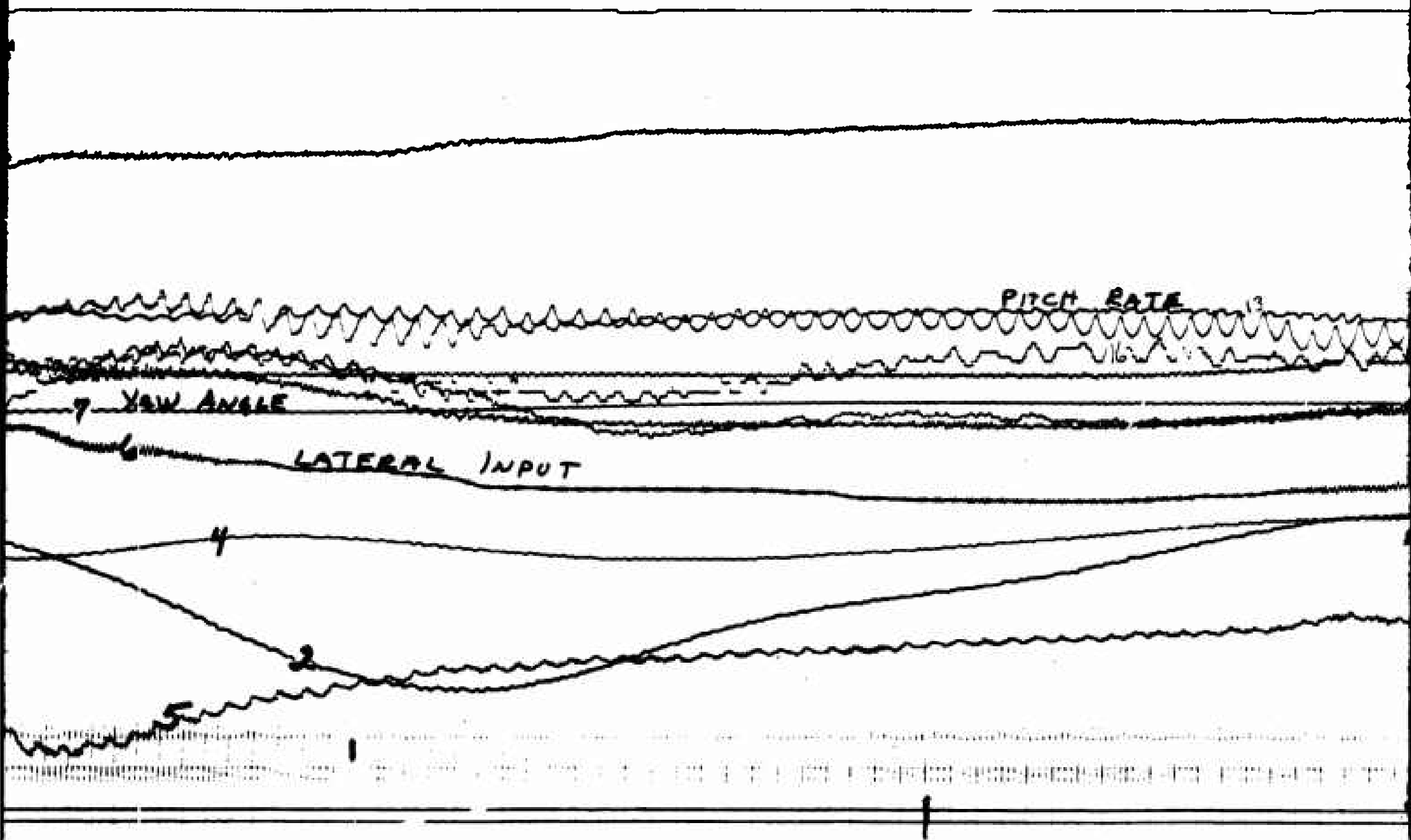


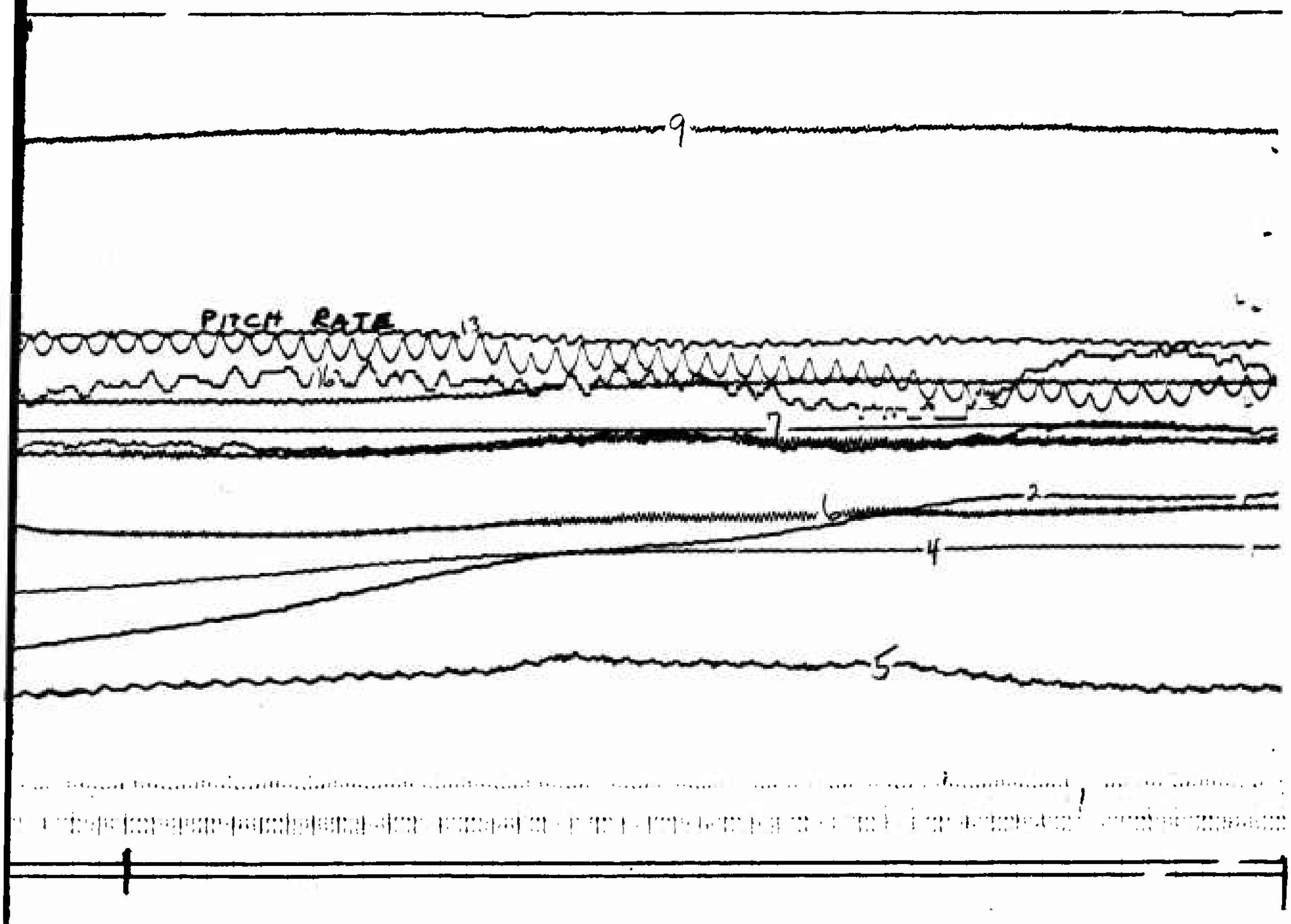
Figure 29. Autorotational Entry - 90 KIAS, 4000 Feet, With Yaw SAS.

H



Feet,

B


9
PITCH RATE
13
16
7
6
2
4
5

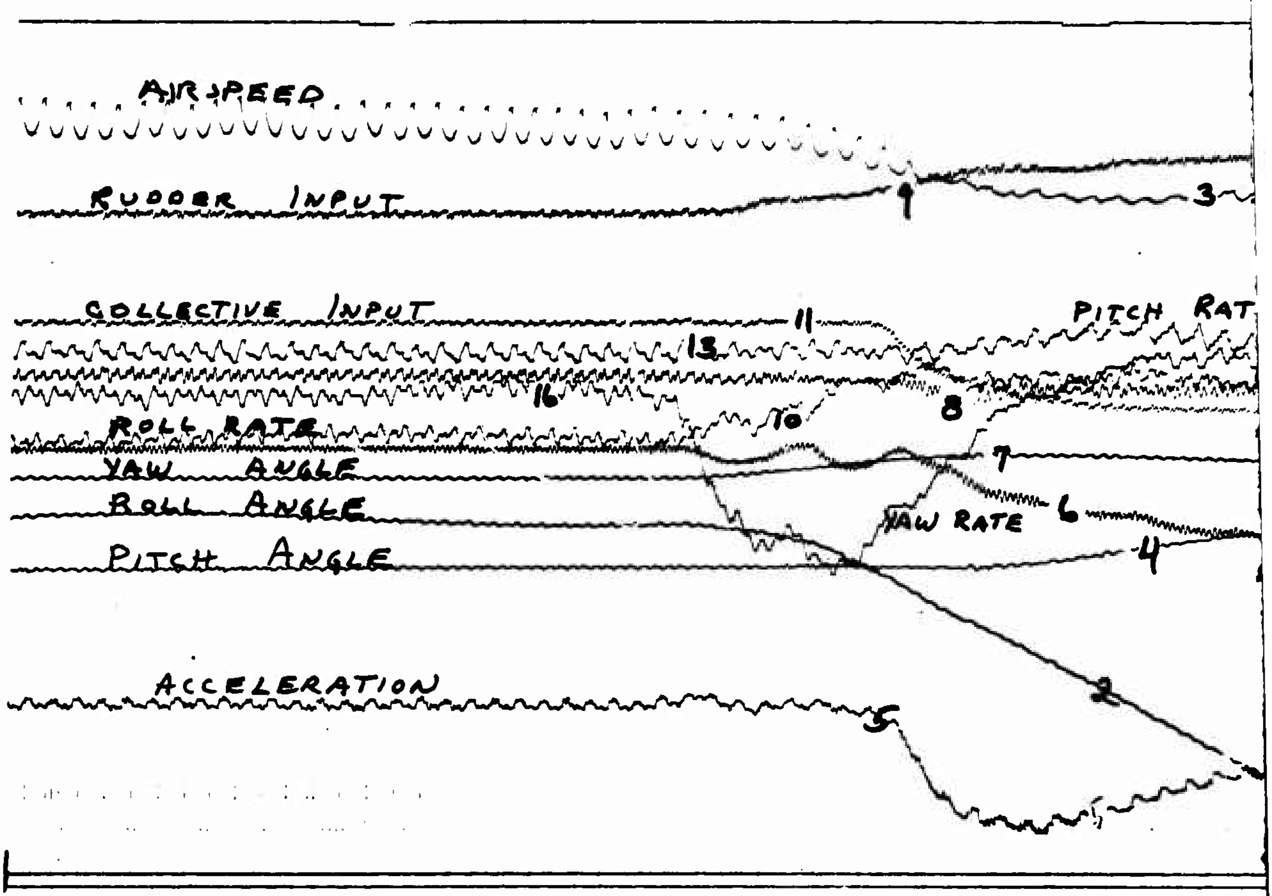
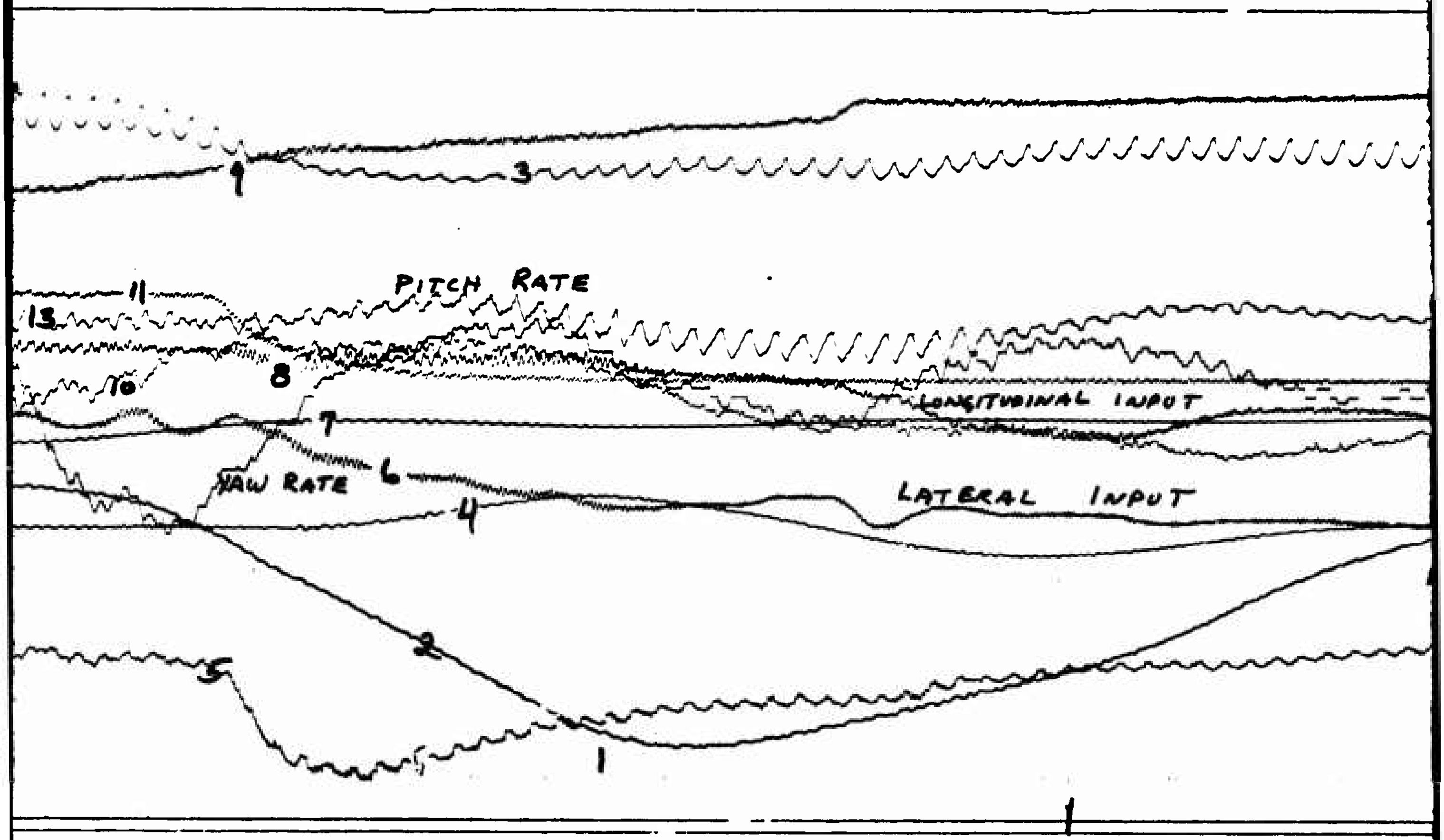


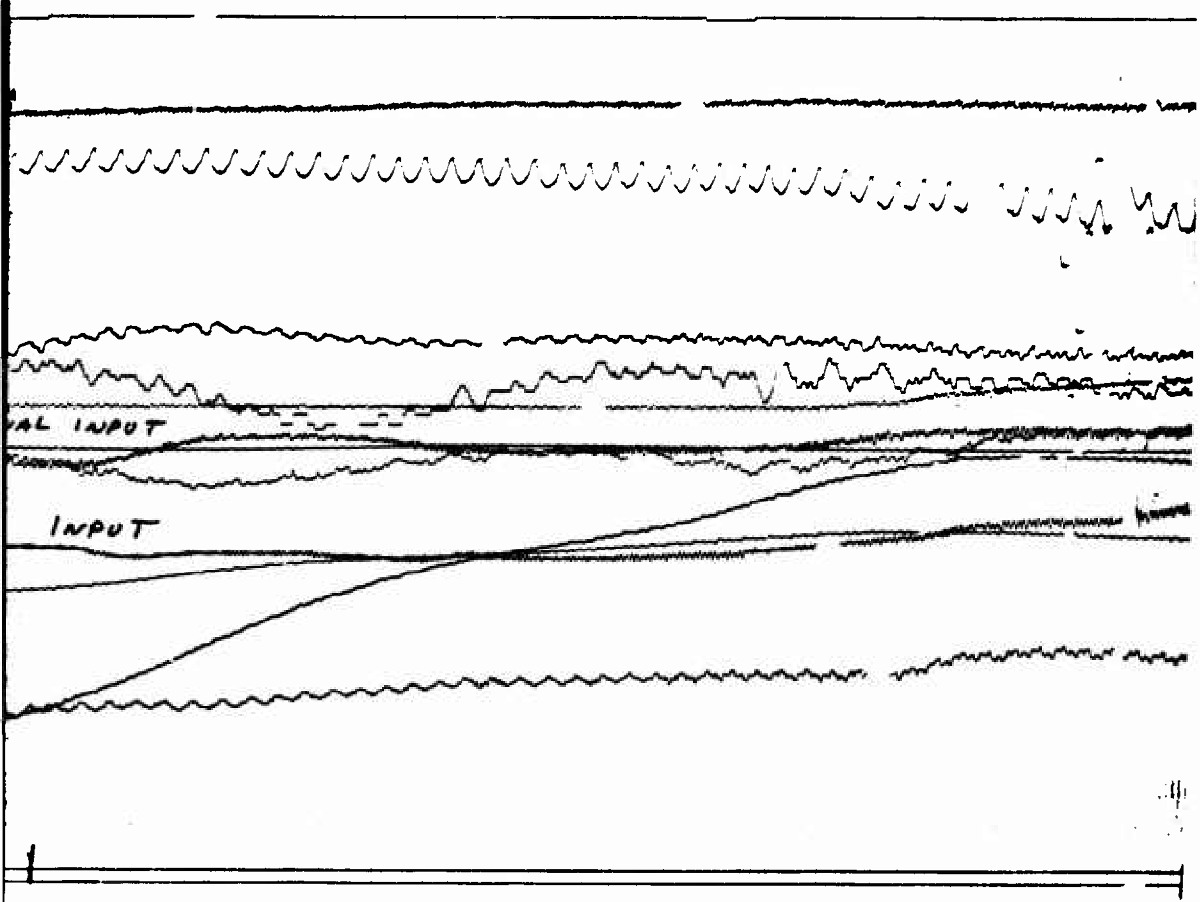
Figure 30. Autorotational Entry - 110 KIAS, 4000 Feet, With Stabilizer Bar.

A



IAS, 4000 Feet,

B


AL INPUT
INPUT

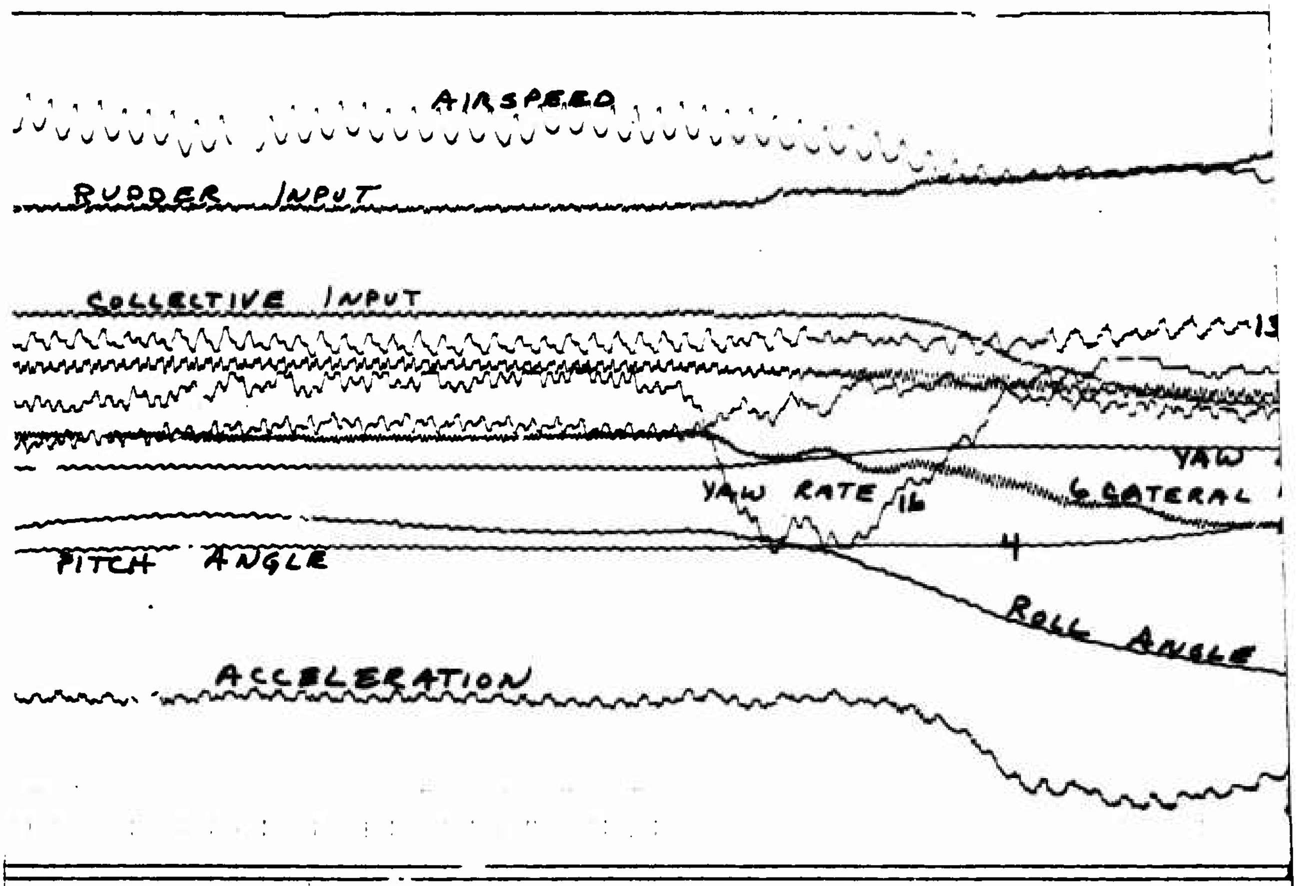
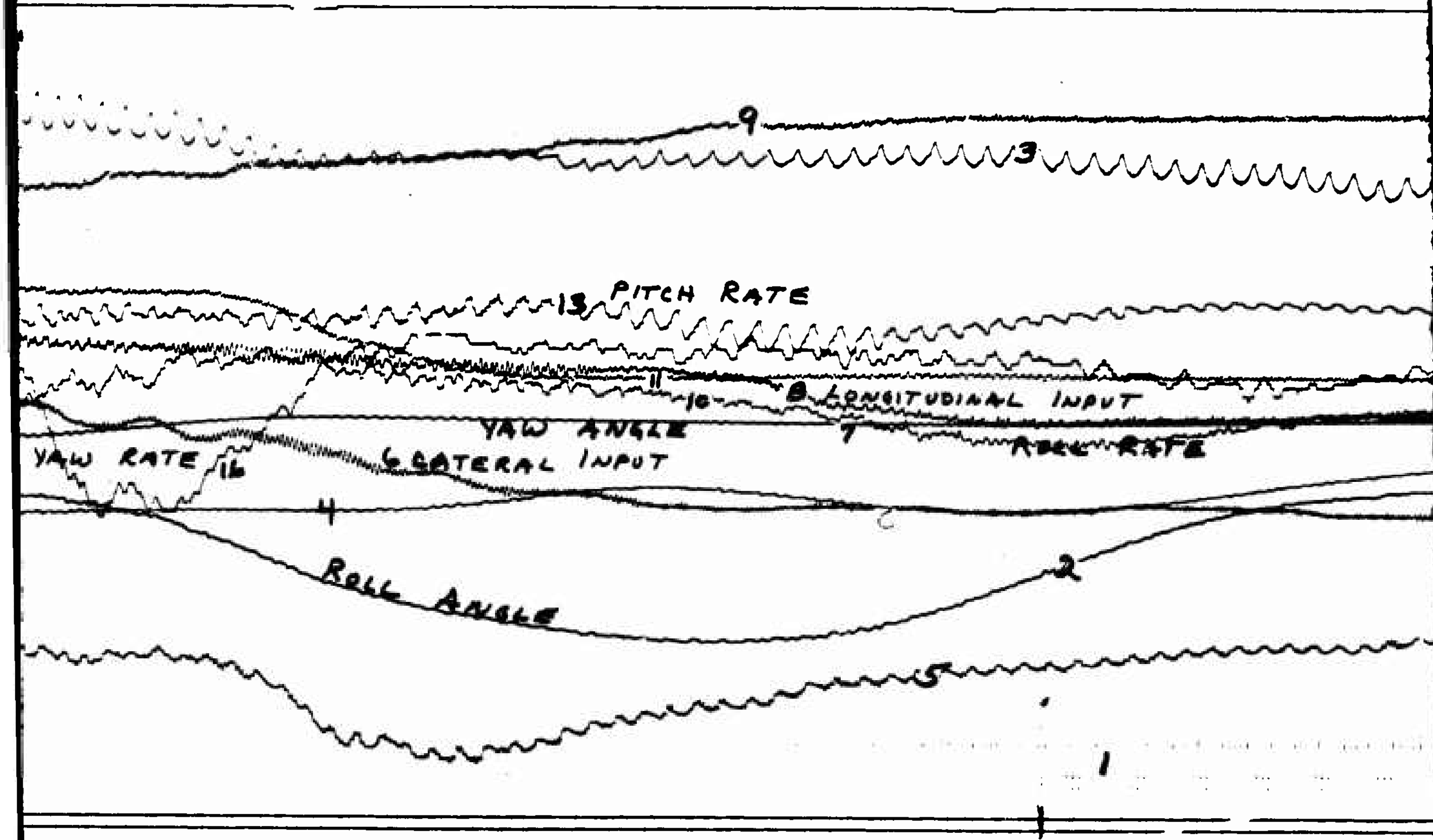


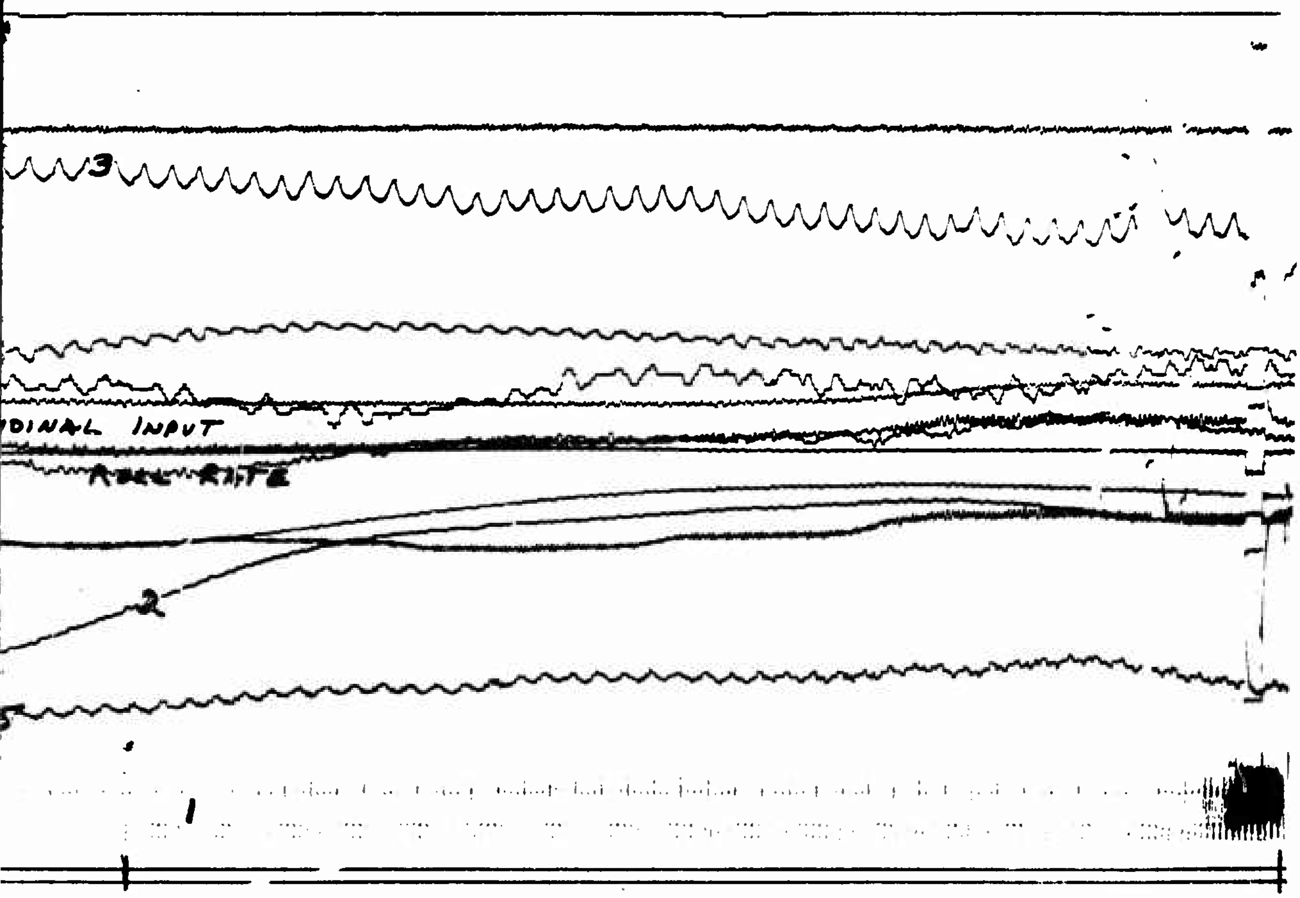
Figure 31. Autorotational Entry - 110 KIAS, 4000 Feet, With Yaw SAS.

A


9
3
13
PITCH RATE
11
10
8 LONGITUDINAL INPUT
YAW ANGLE
7
ROLL RATE
YAW RATE
16
6 LATERAL INPUT
4
ROLL ANGLE
2
5
1

, 4000 Feet,

12



C

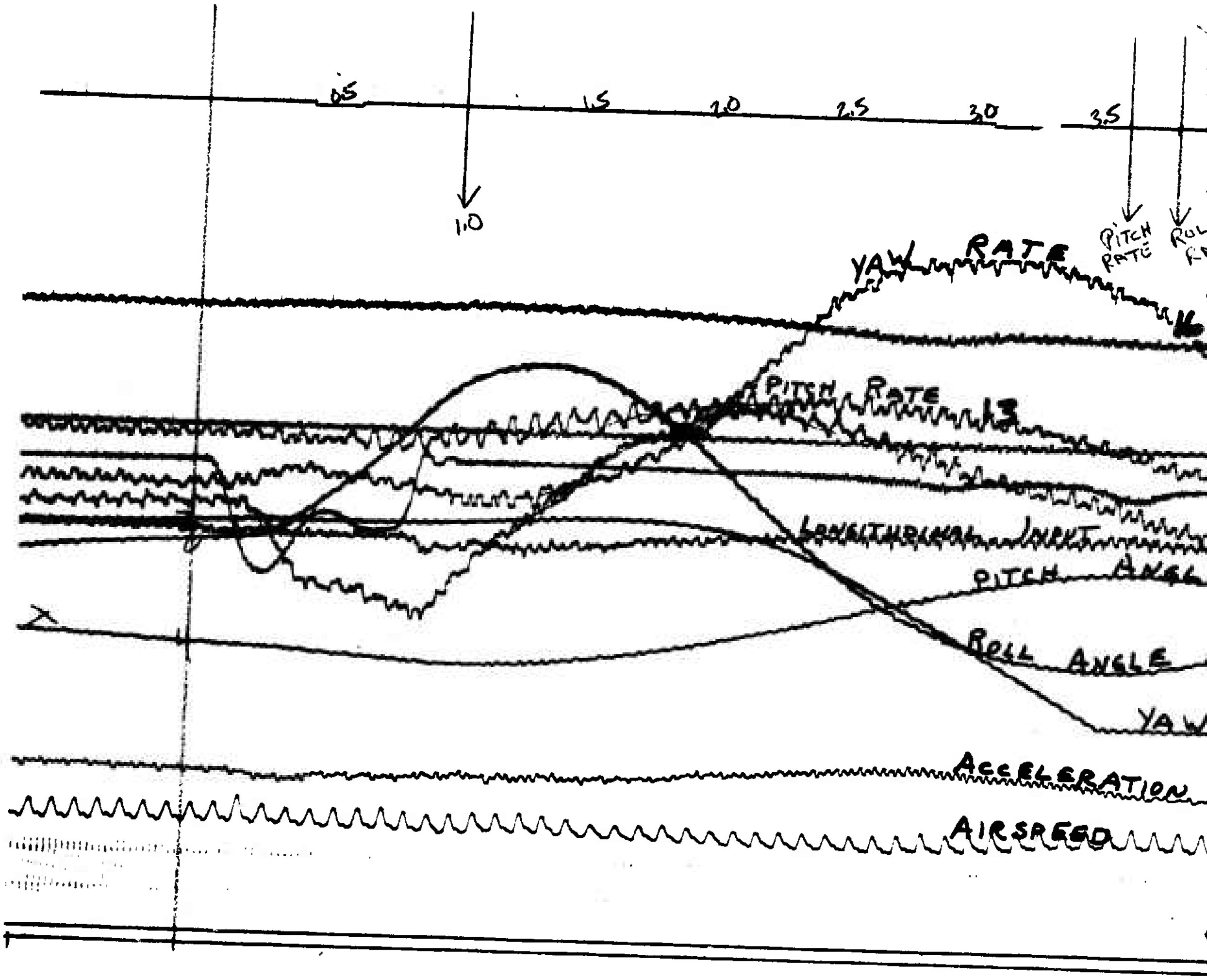
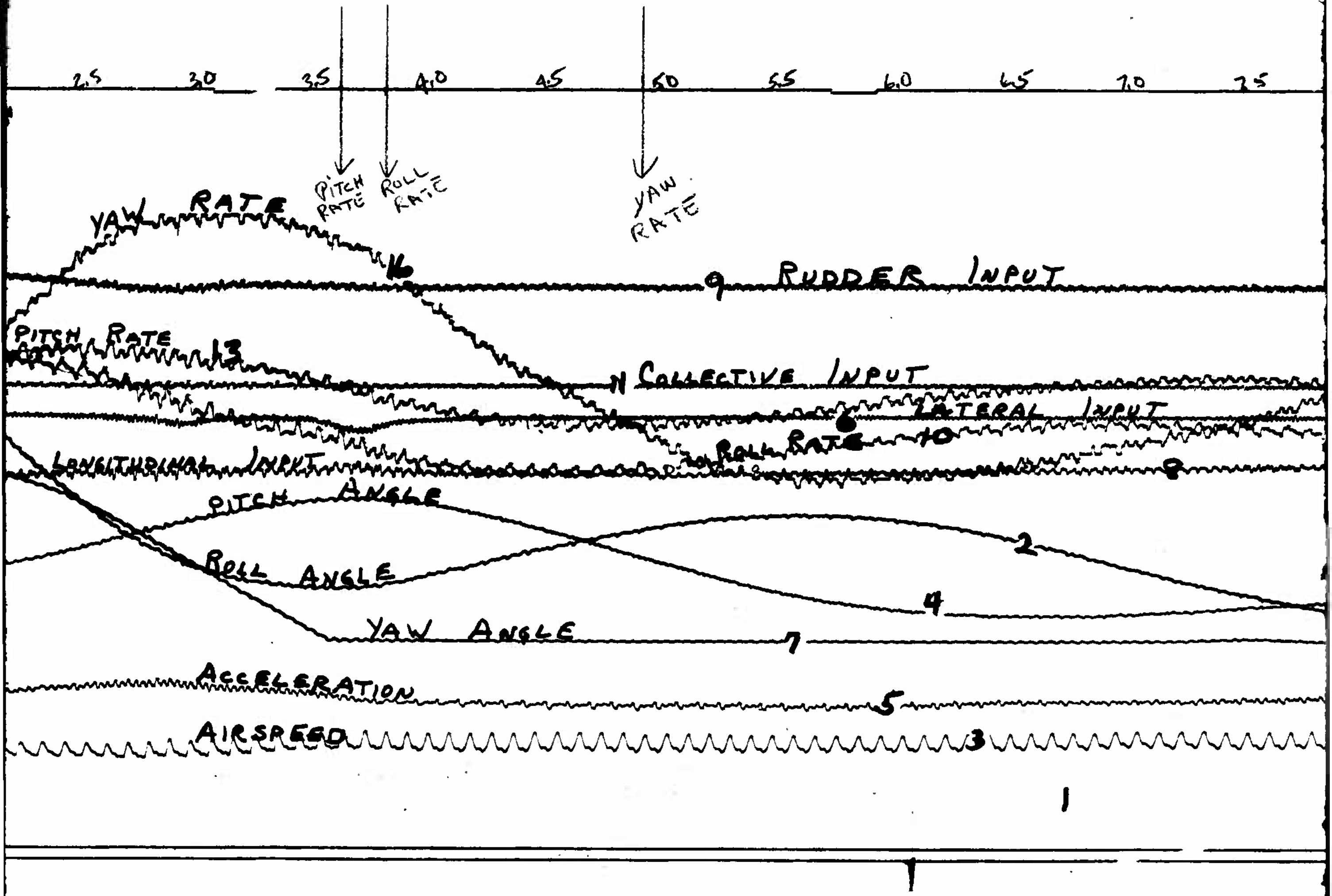


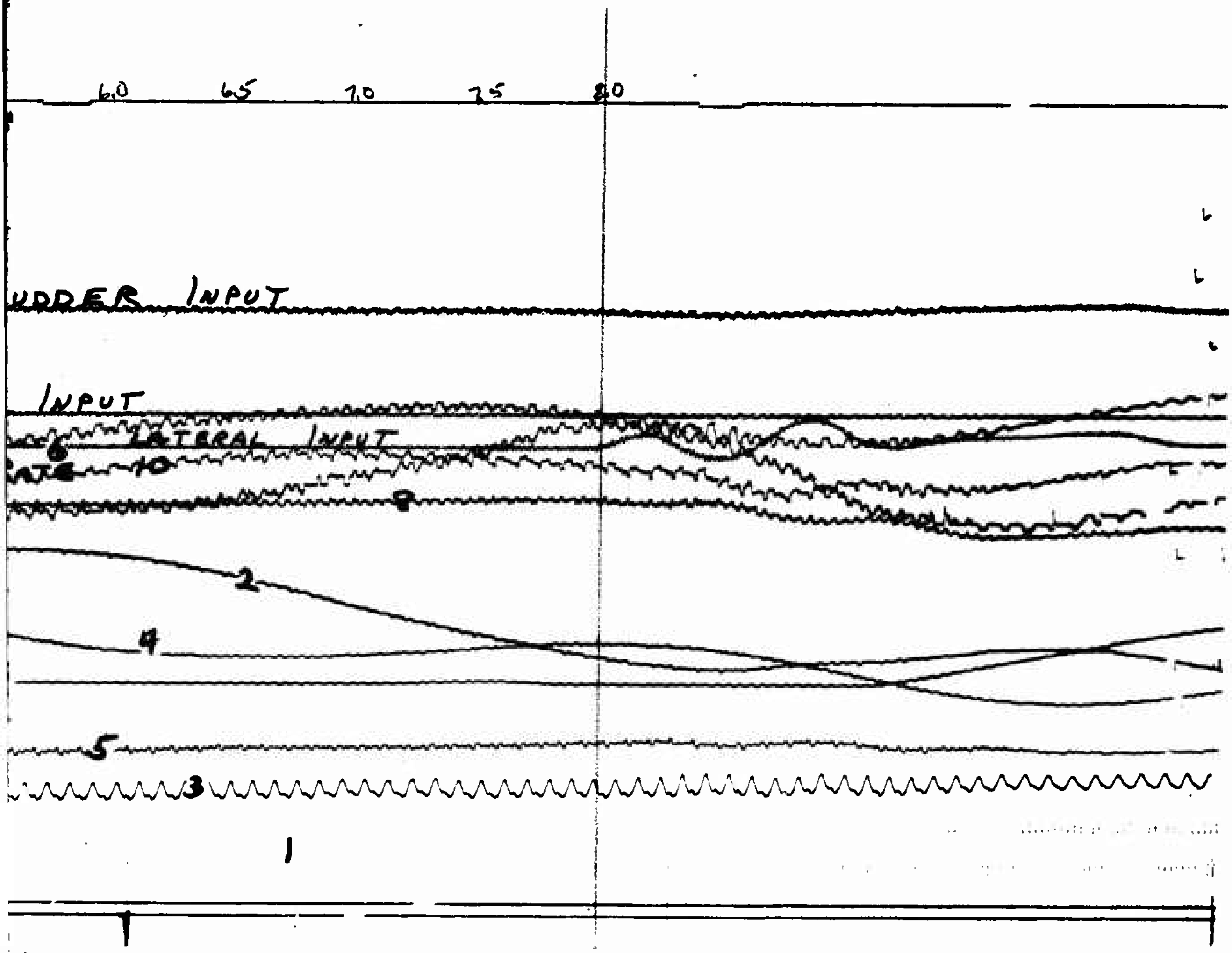
Figure 32. Right Lateral Pulse - Hover, With Stabilizer Bar.

A



2.5
3.0
3.5
4.0
4.5
5.0
5.5
6.0
6.5
7.0
7.5
PITCH RATE
ROLL RATE
YAW RATE
YAW RATE 16
9 RUDDER INPUT
PITCH RATE 13
11 COLLECTIVE INPUT
LATERAL INPUT
6
ROLL RATE 10
LONGITUDINAL INPUT
8
PITCH ANGLE 2
ROLL ANGLE 4
YAW ANGLE 7
ACCELERATION 5
AIRSPEED 3
1

B



6.0
6.5
7.0
7.5
8.0
UDDER INPUT
INPUT
LATERAL INPUT
6
ATE
10
8
2
4
5
3
1

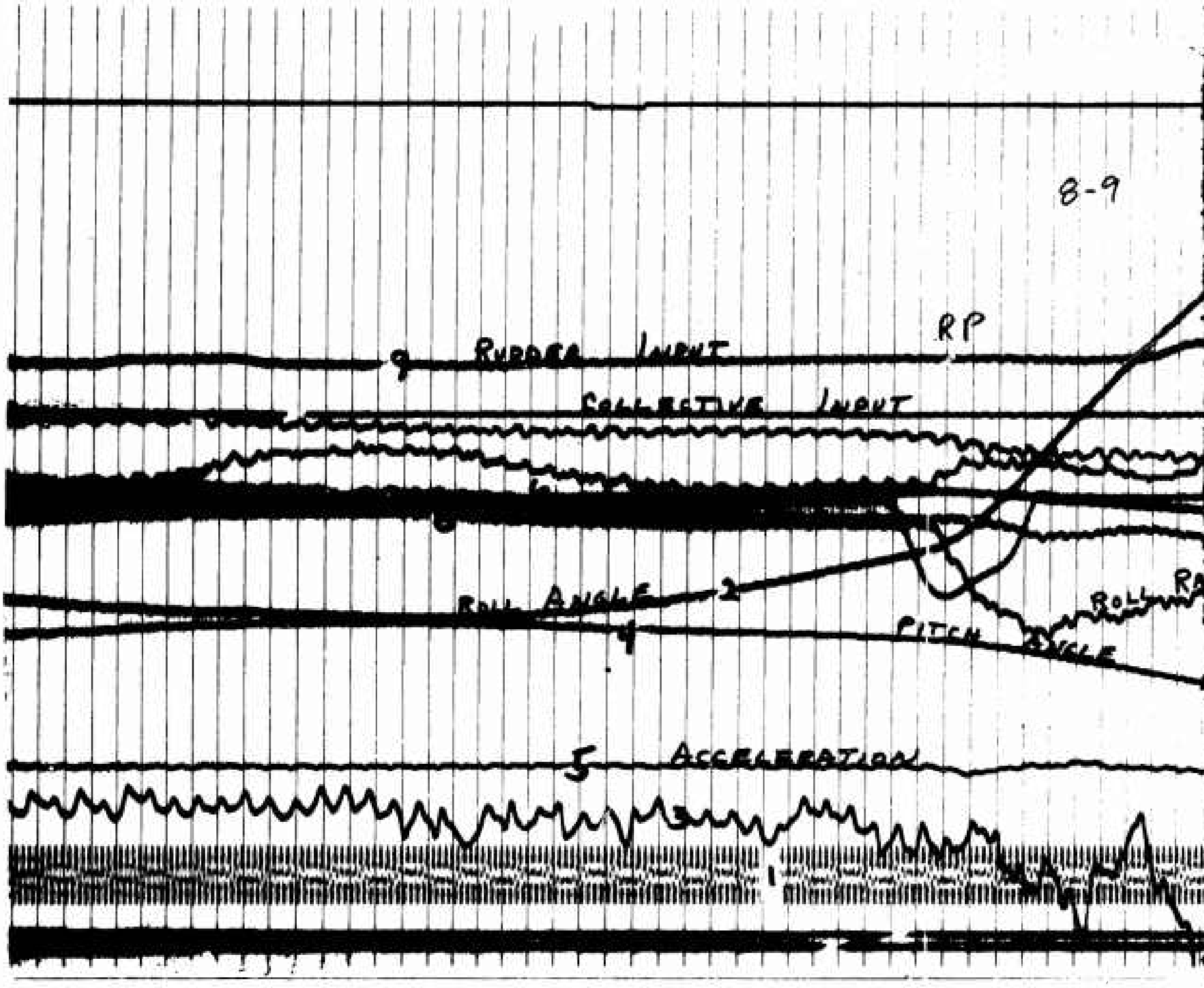
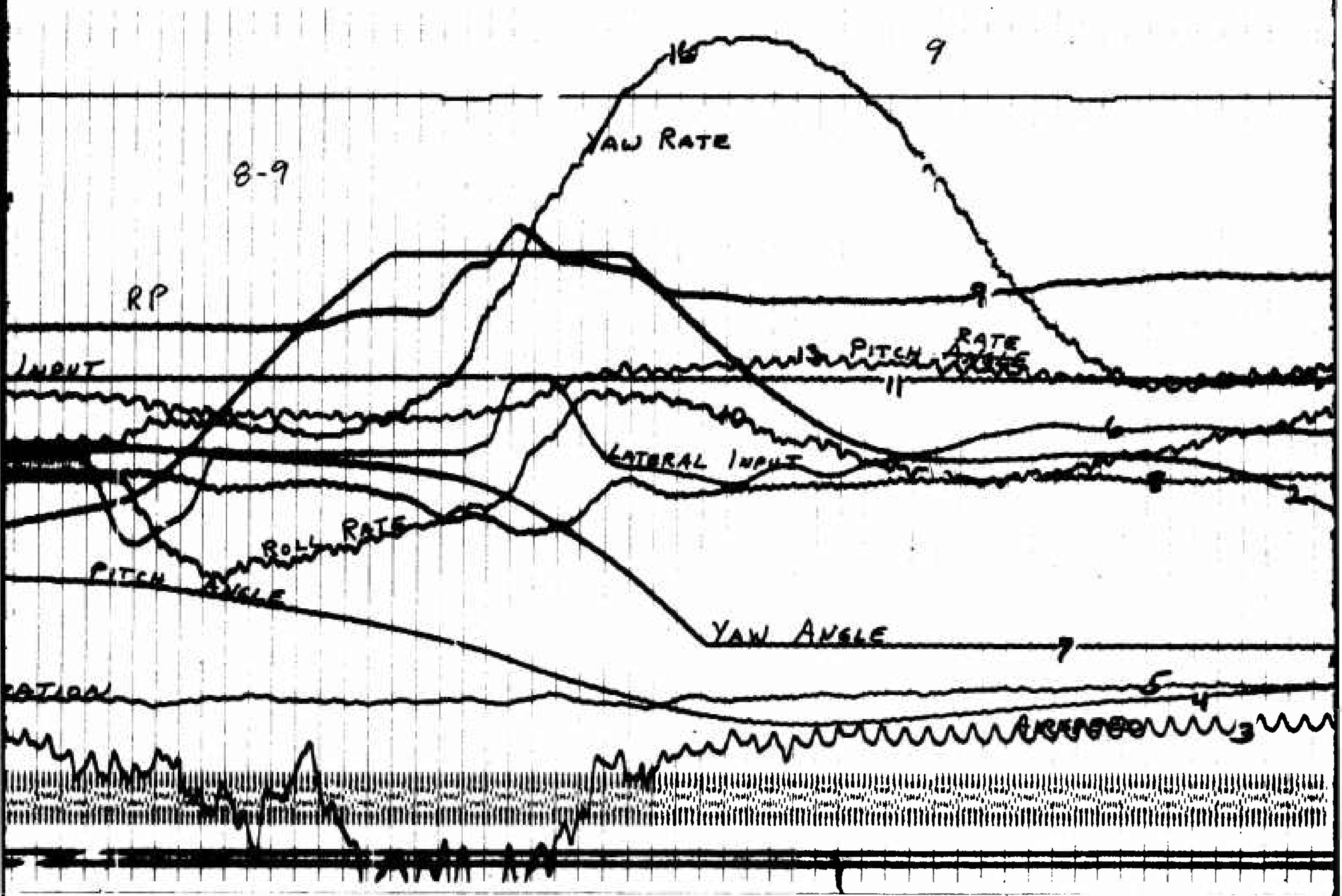


Figure 33. Right Lateral Pulse - Hover, Without Stabilizer Bar and SAS.

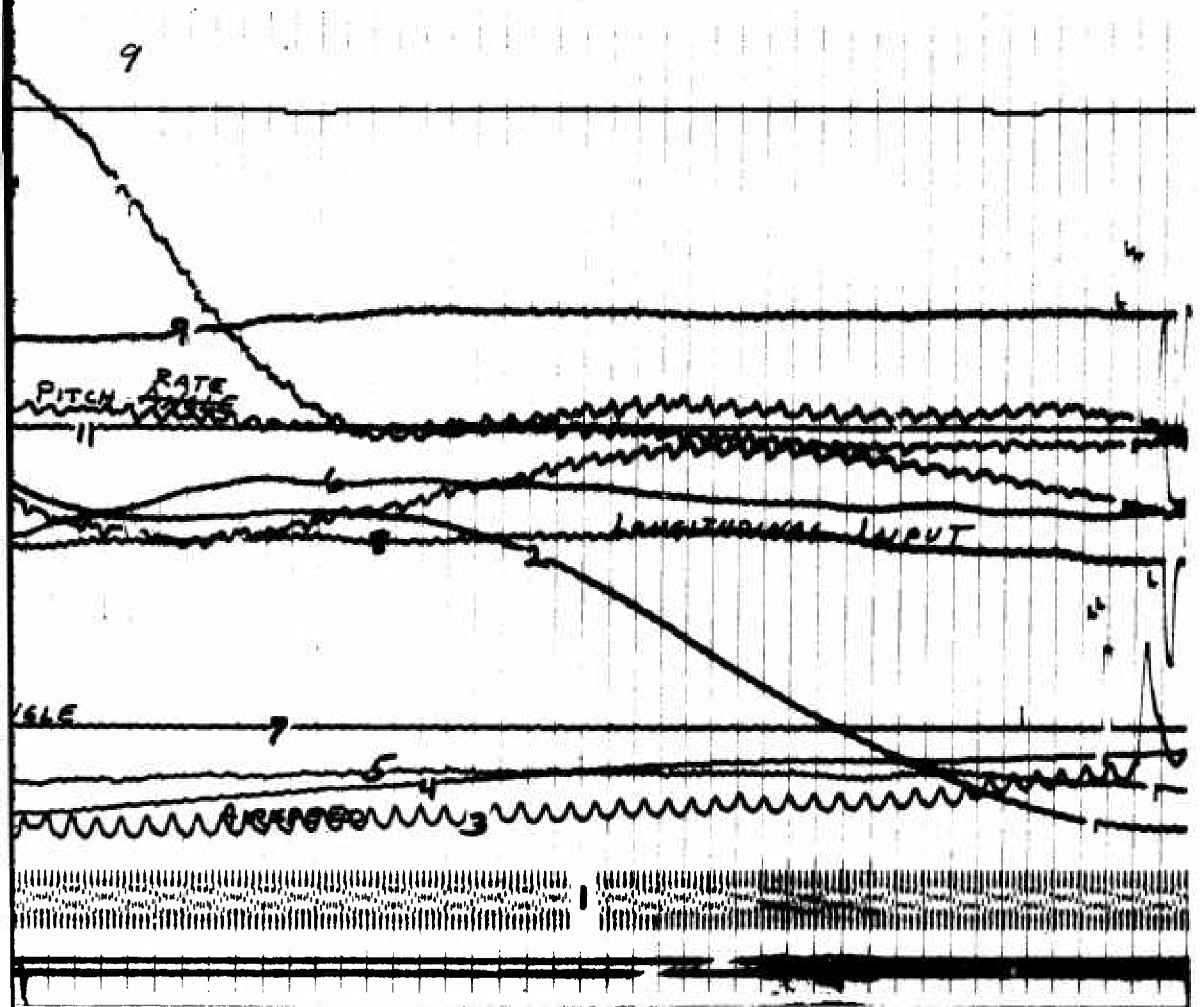
A



16
9
YAW RATE
8-9
RP
9
INPUT
13 PITCH RATE
11
10
6
LATERAL INPUT
8
2
ROLL RATE
PITCH ANGLE
YAW ANGLE
7
RATION
5
4
3

ıt

13



C

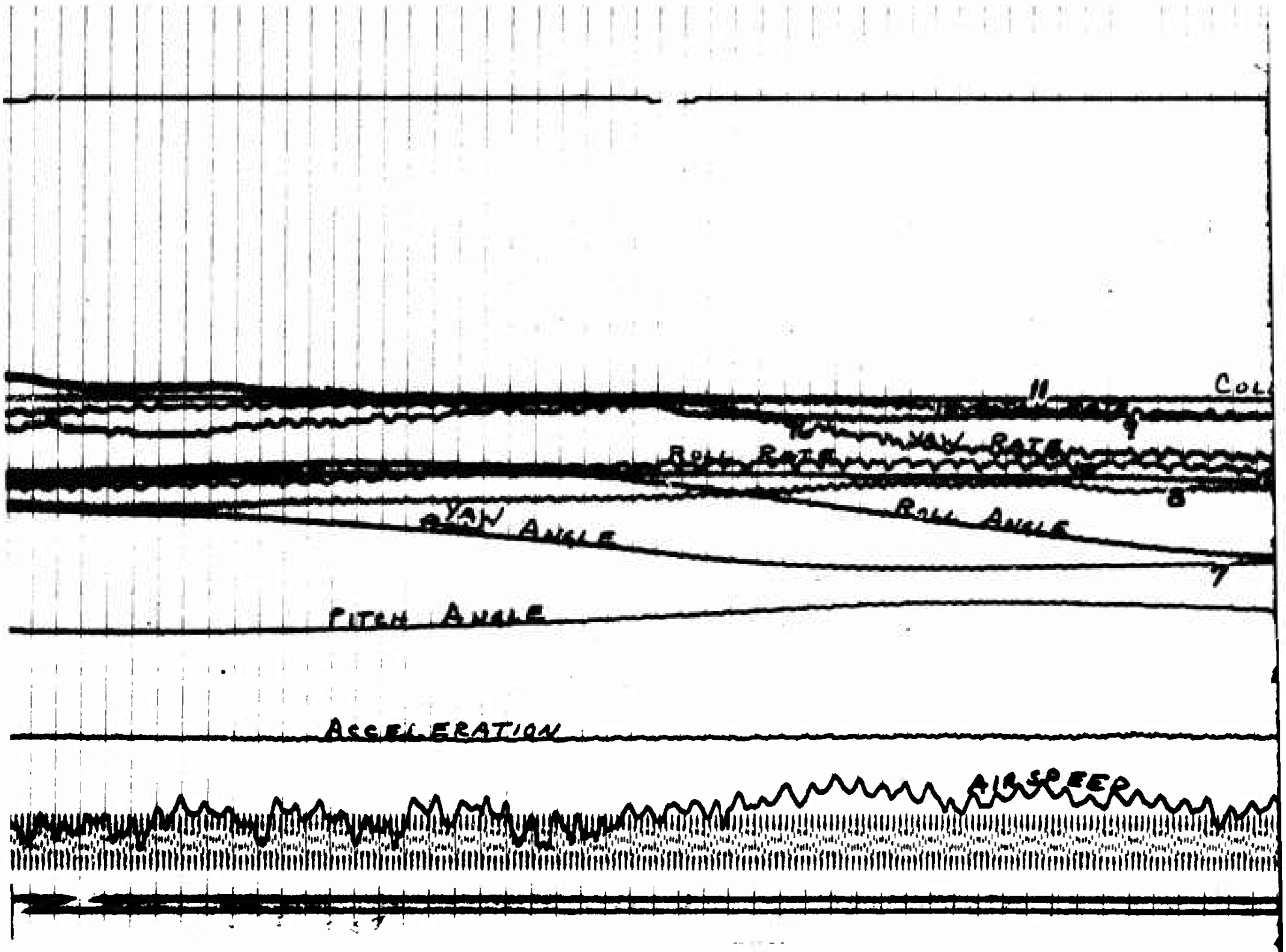
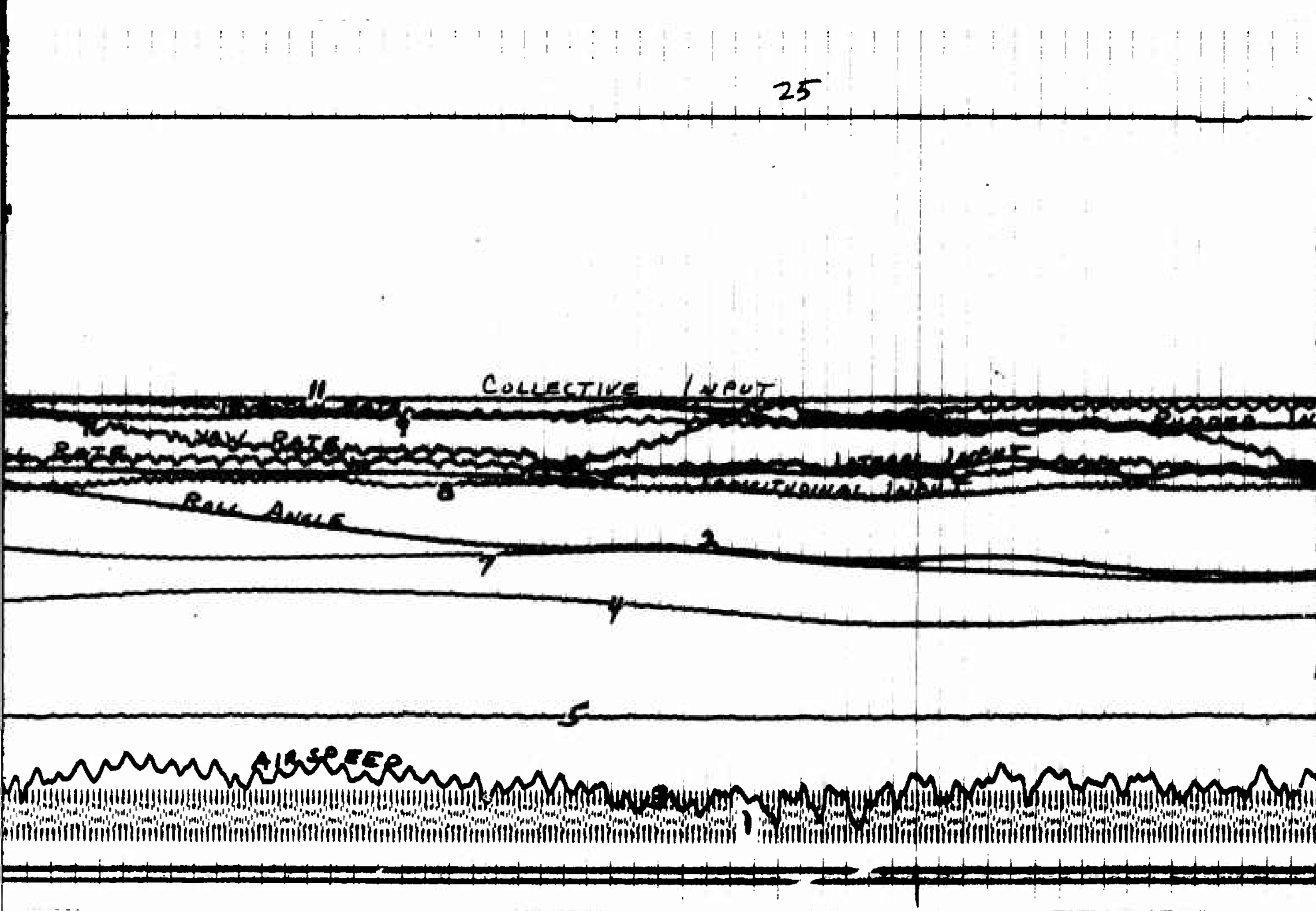
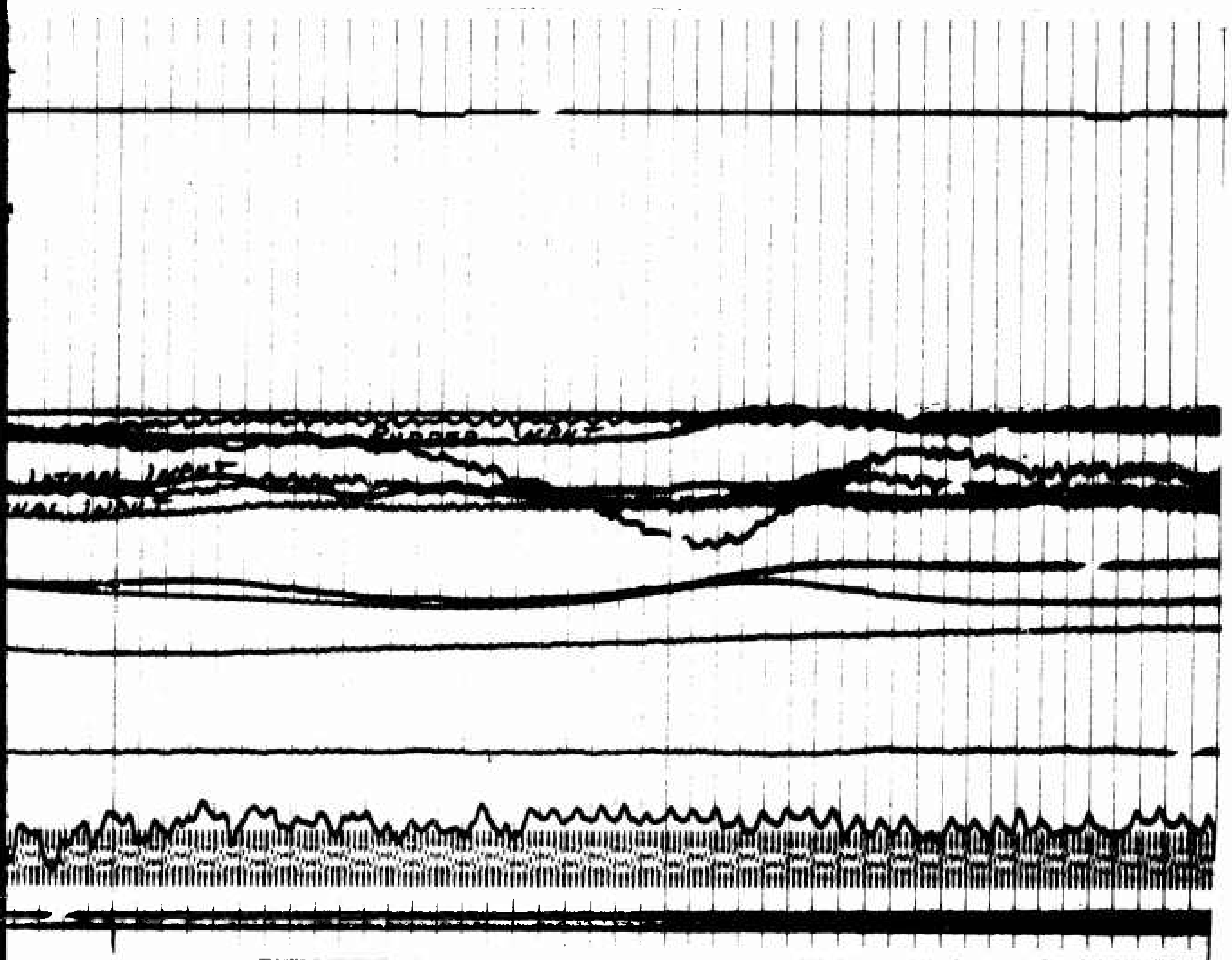


Figure 34. Right Lateral Pulse - Hover, With MSAS.

A


25
COLLECTIVE INPUT
YAW RATE
ROLL ANGLE
AIRSPEED



C

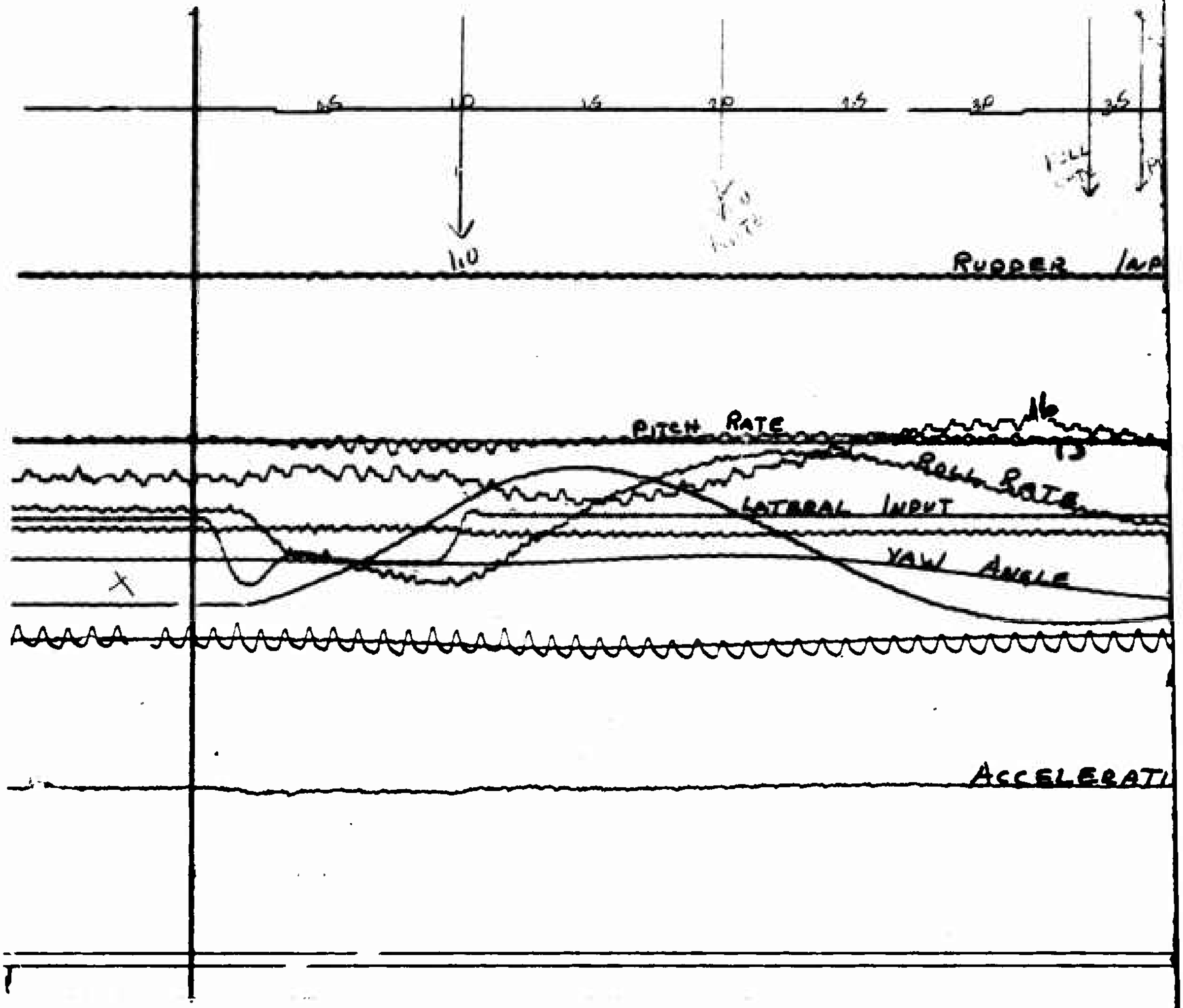
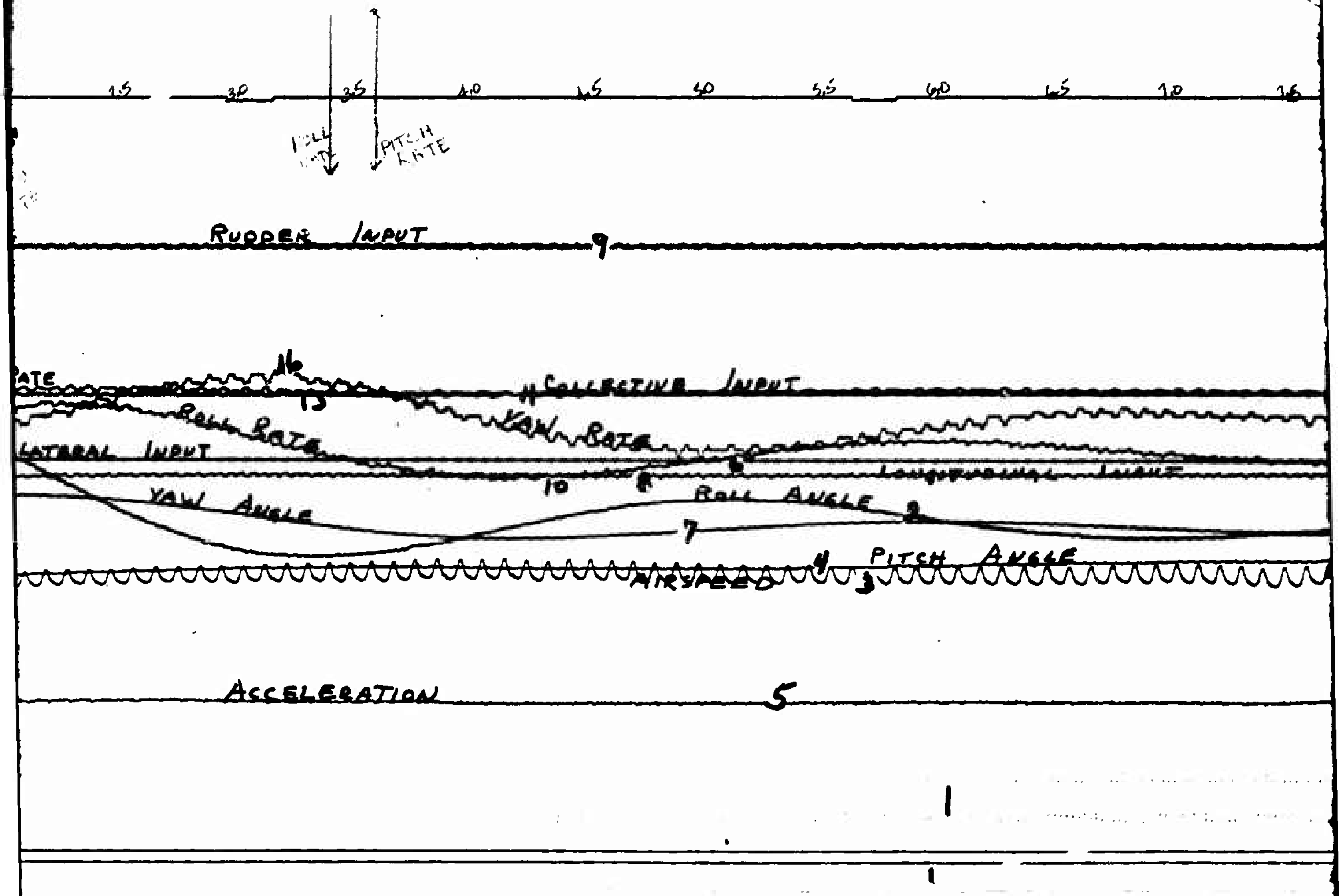
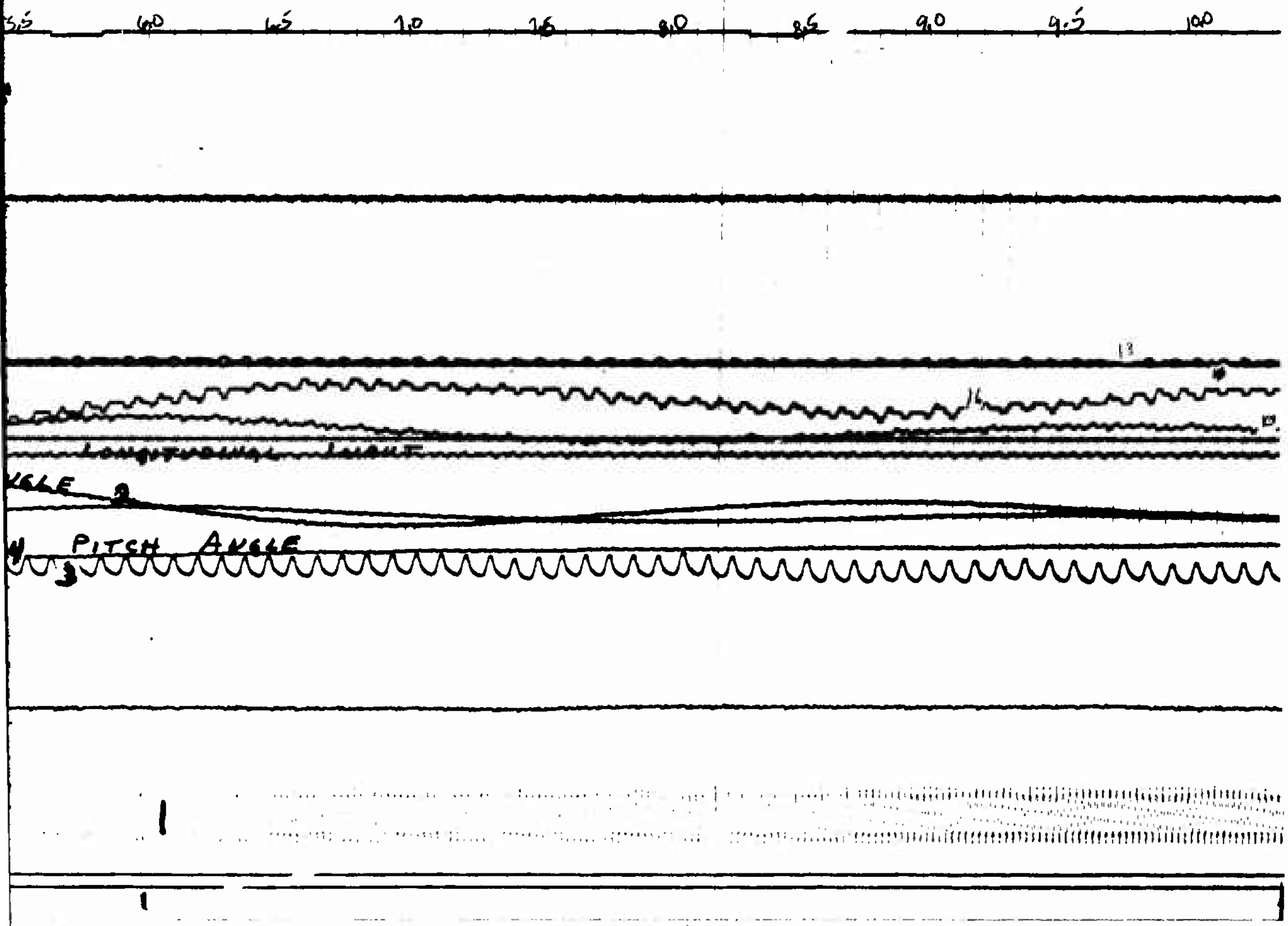


Figure 35. Right Lateral Pulse - 60 KIAS, 4000 Feet,
With Stabilizer Bar.


30
35
40
45
50
55
60
65
70
75
PITCH RATE
RUDDER INPUT
9
16
13
11
COLLECTIVE INPUT
ROLL RATE
YAW RATE
LATERAL INPUT
10
8
6
LONGITUDINAL INPUT
YAW ANGLE
ROLL ANGLE
2
7
4
PITCH ANGLE
AIRSPEED
3
ACCELERATION
5
1

0 Feet,



C

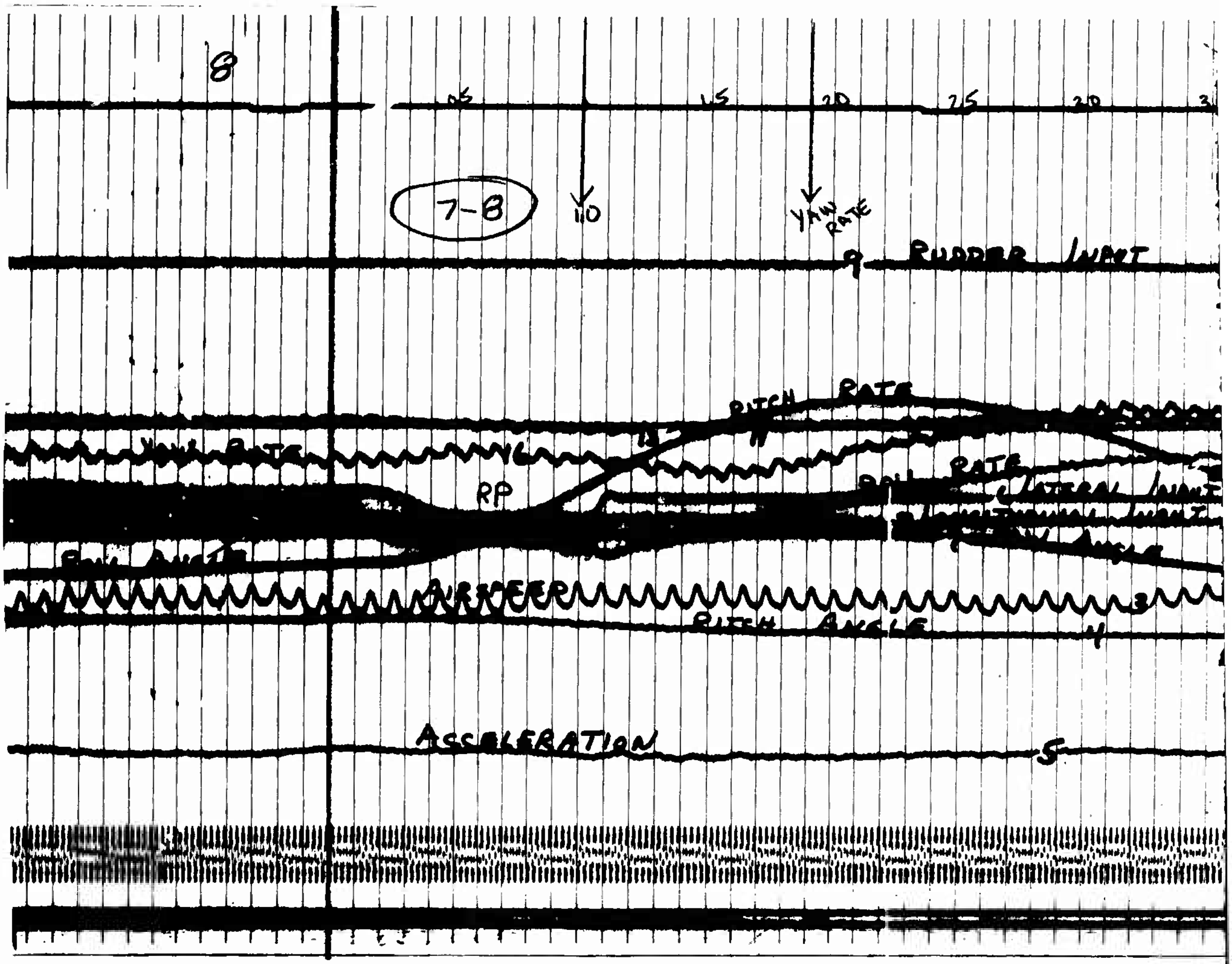
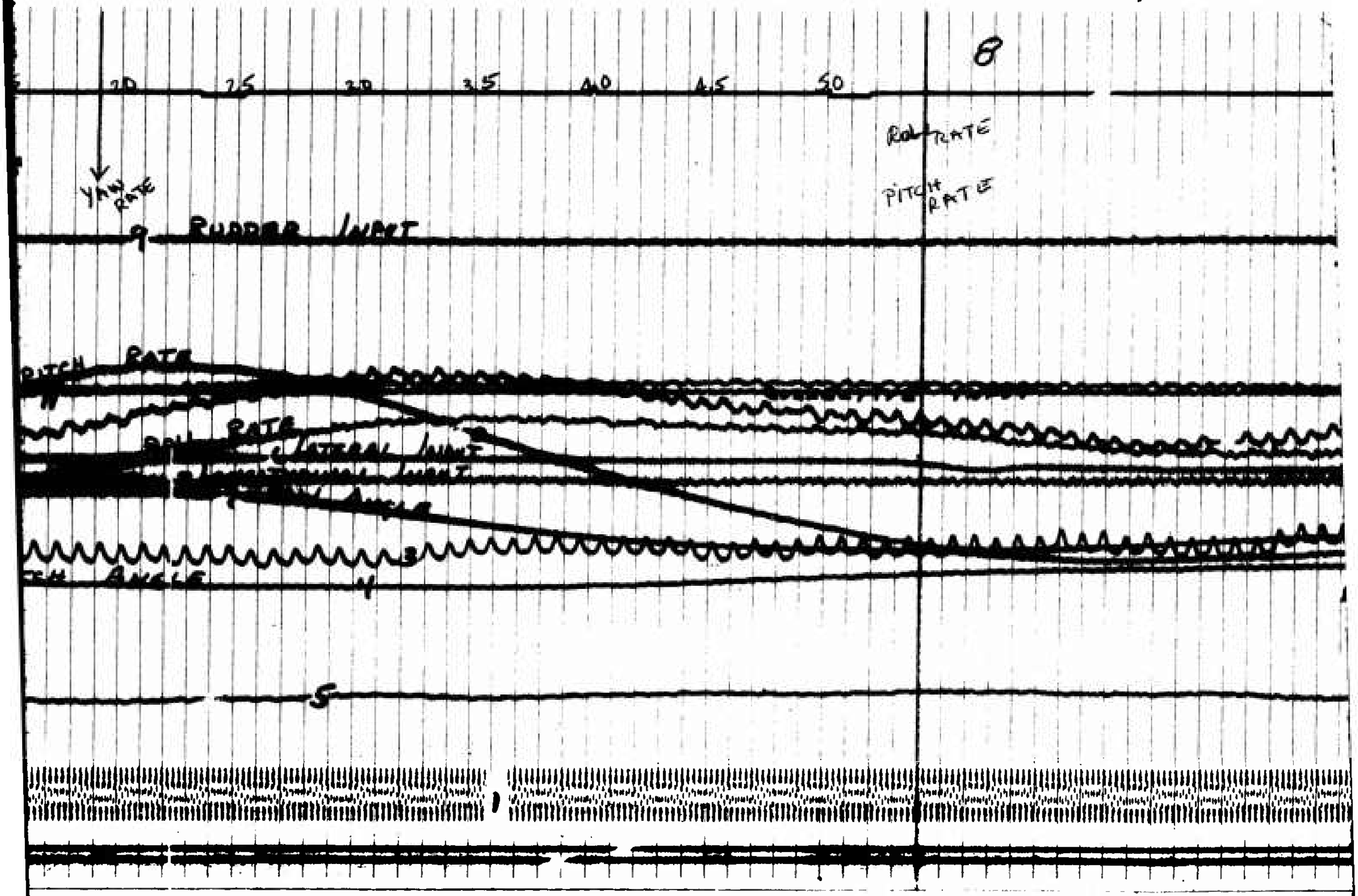


Figure 36. Right Lateral Pulse - 60 KIAS, 4000 Feet,
Without Stabilizer Bar and SAS.

H



20
25
30
35
40
45
50
8
ROLL RATE
PITCH RATE
YAW RATE
RUDDER INPUT
PITCH RATE
ROLL RATE
LATERAL INPUT
ANGLE
ANGLE

Feet,

C

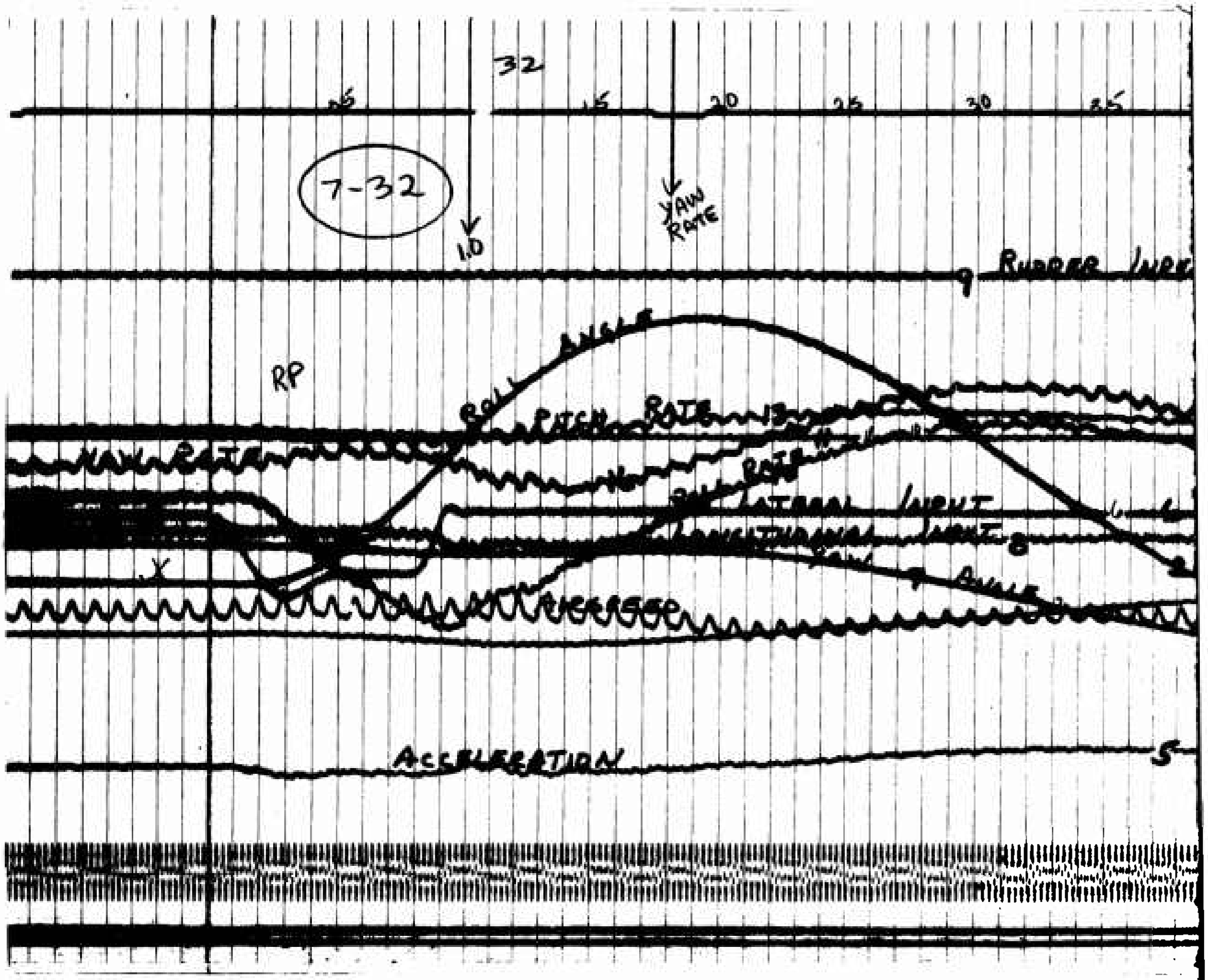
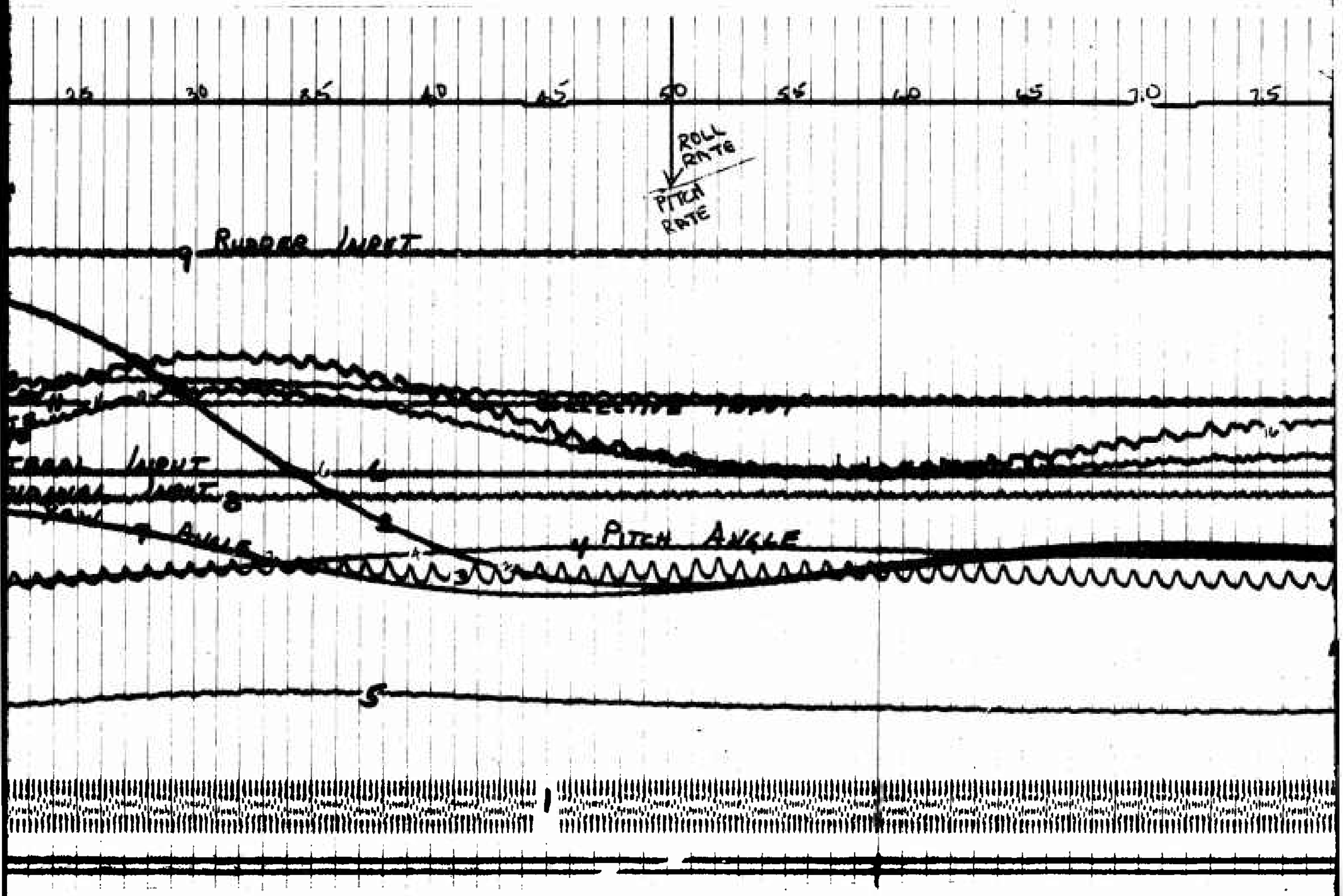
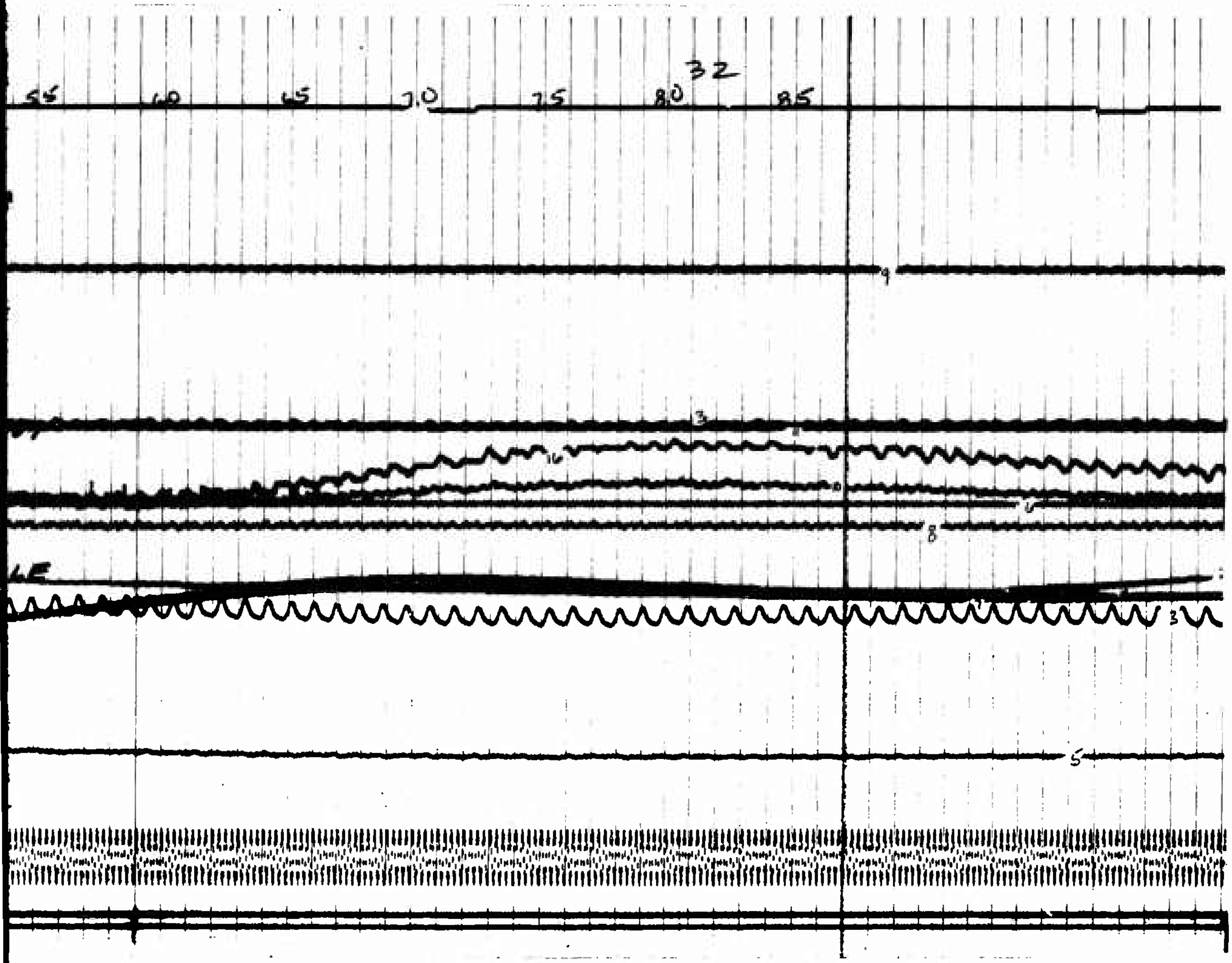


Figure 37. Right Lateral Pulse - 60 KIAS, 4000 Feet, With MSAS.

A


25
30
35
40
45
50
55
60
65
70
75
ROLL RATE
PITCH RATE
RUDDER INPUT
PITCH ANGLE



C

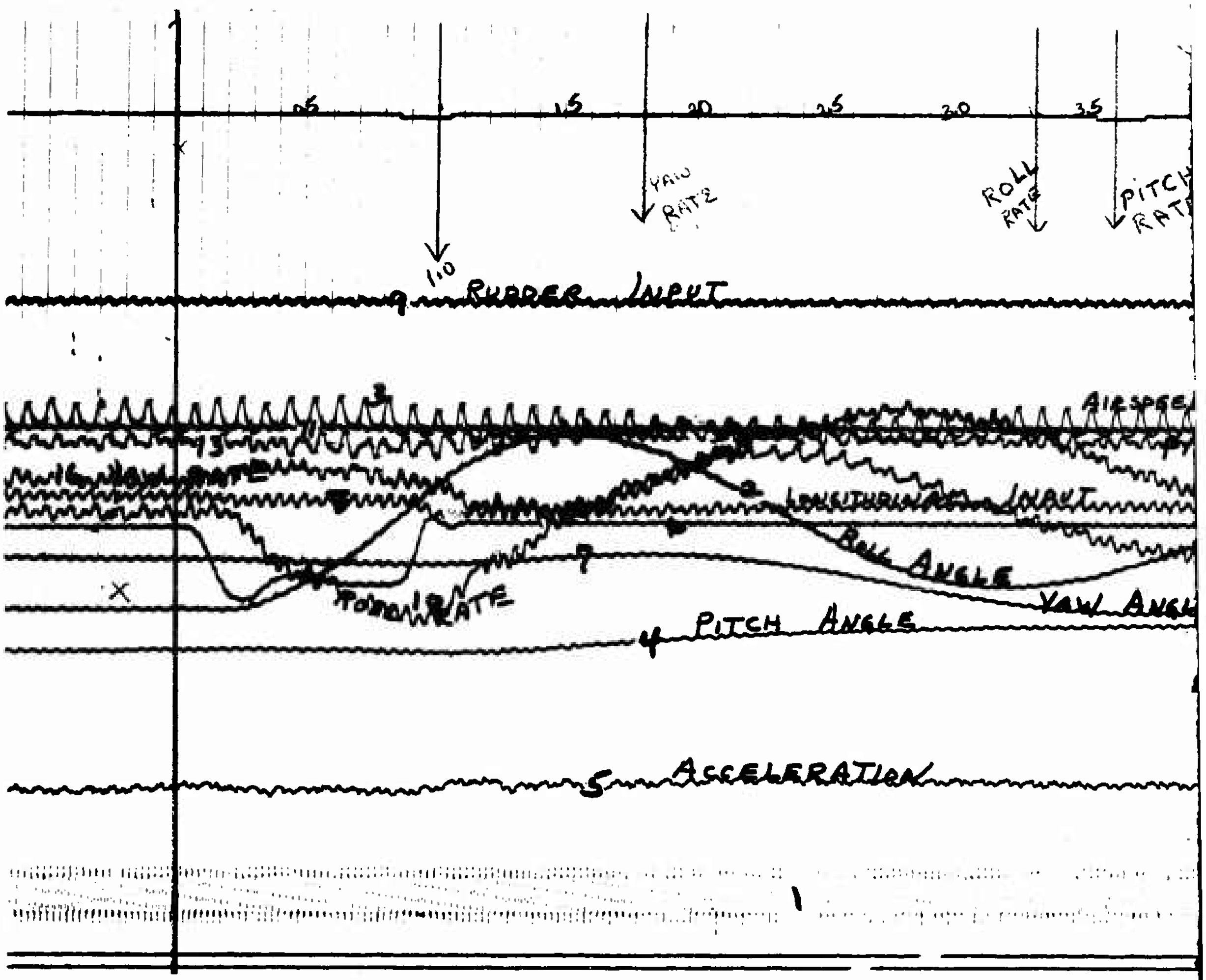
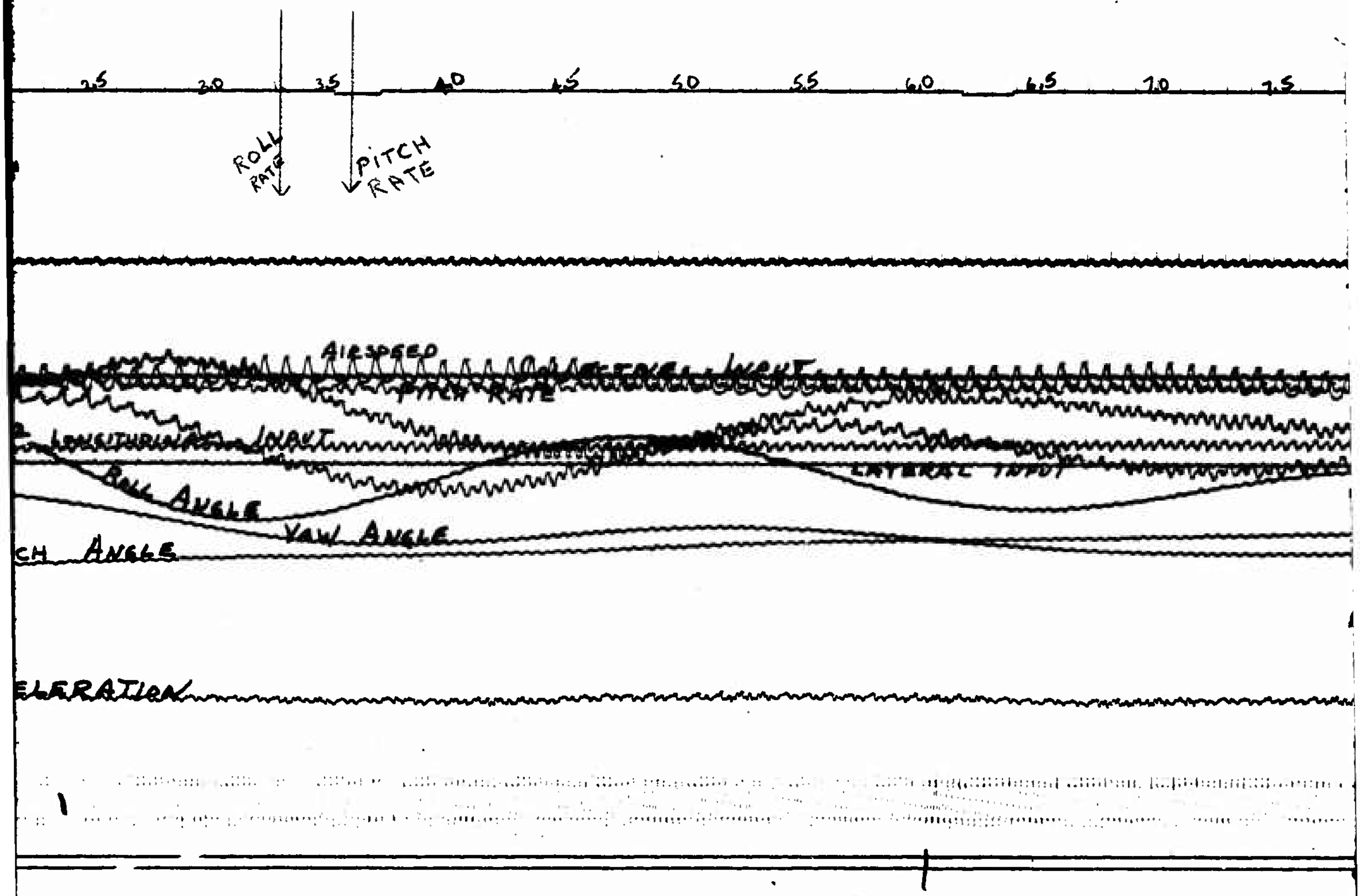


Figure 38. Right Lateral Pulse - 90 KIAS, 4000 Feet, With Stabilizer Bar.

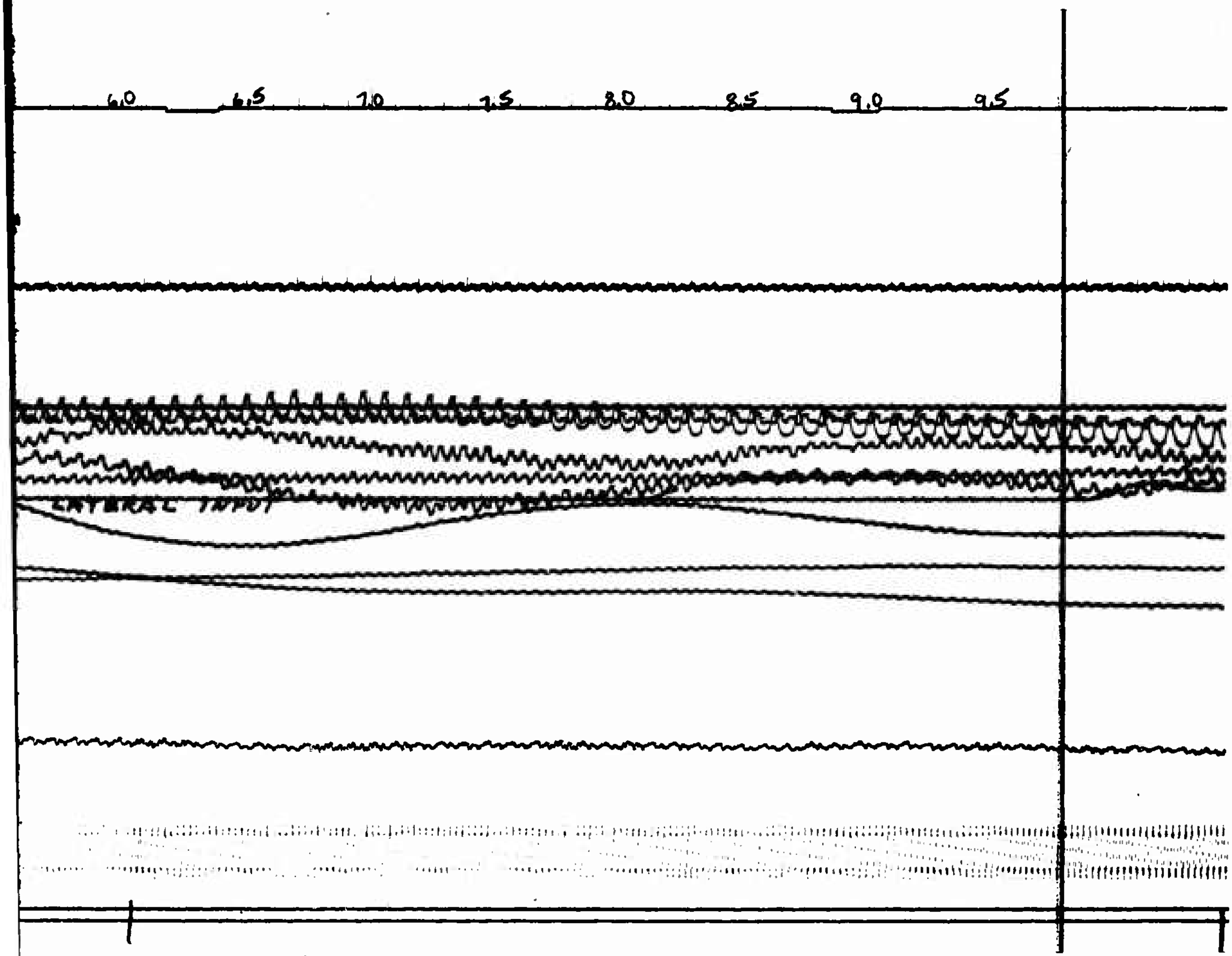
A



2.5
3.0
3.5
4.0
4.5
5.0
5.5
6.0
6.5
7.0
7.5
ROLL RATE
PITCH RATE
AIRSPEED
PITCH RATE
LONGITUDINAL INPUT
ROLL ANGLE
YAW ANGLE
LATERAL INPUT
CH ANGLE
ELERATION

0 Feet,

B



C

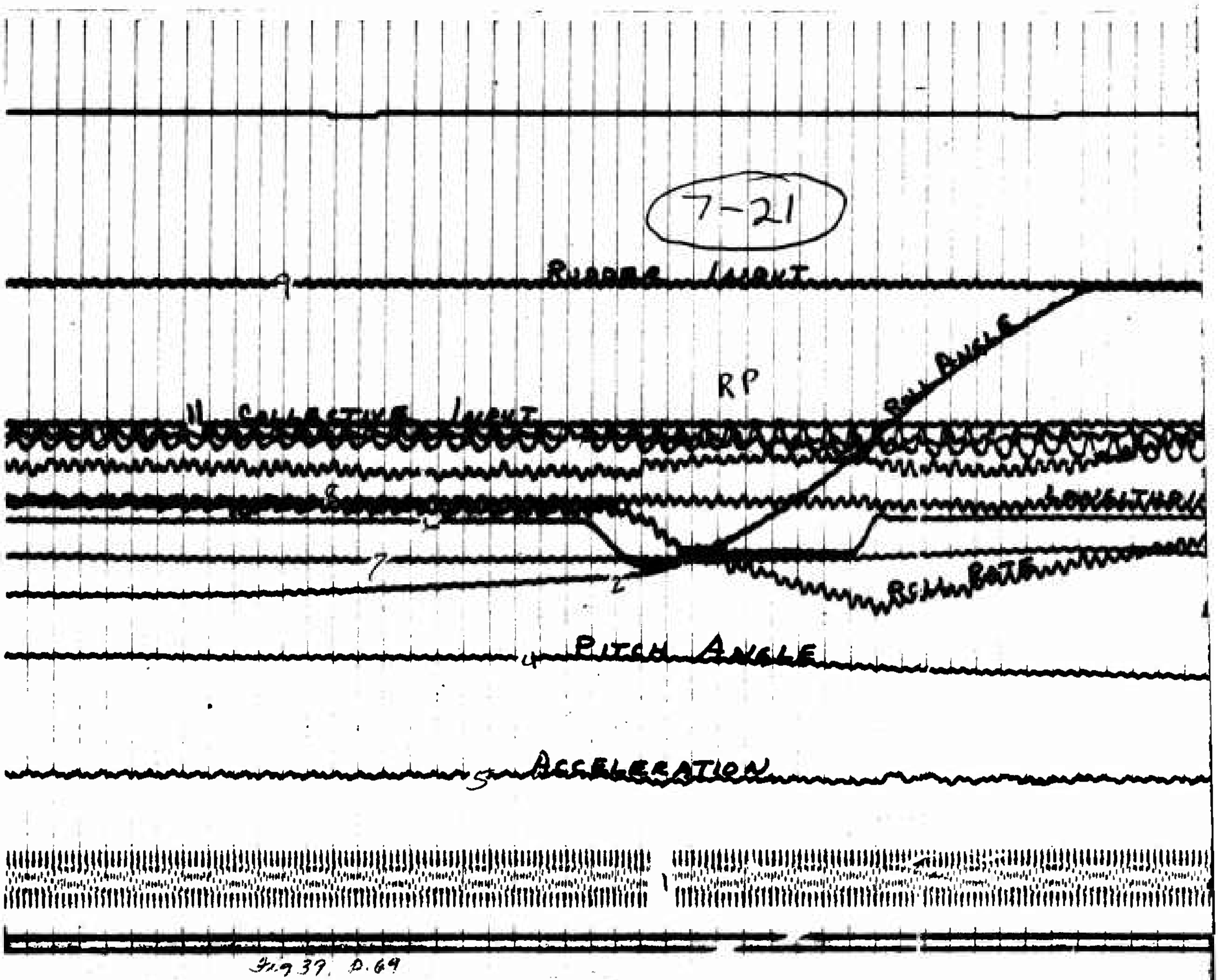
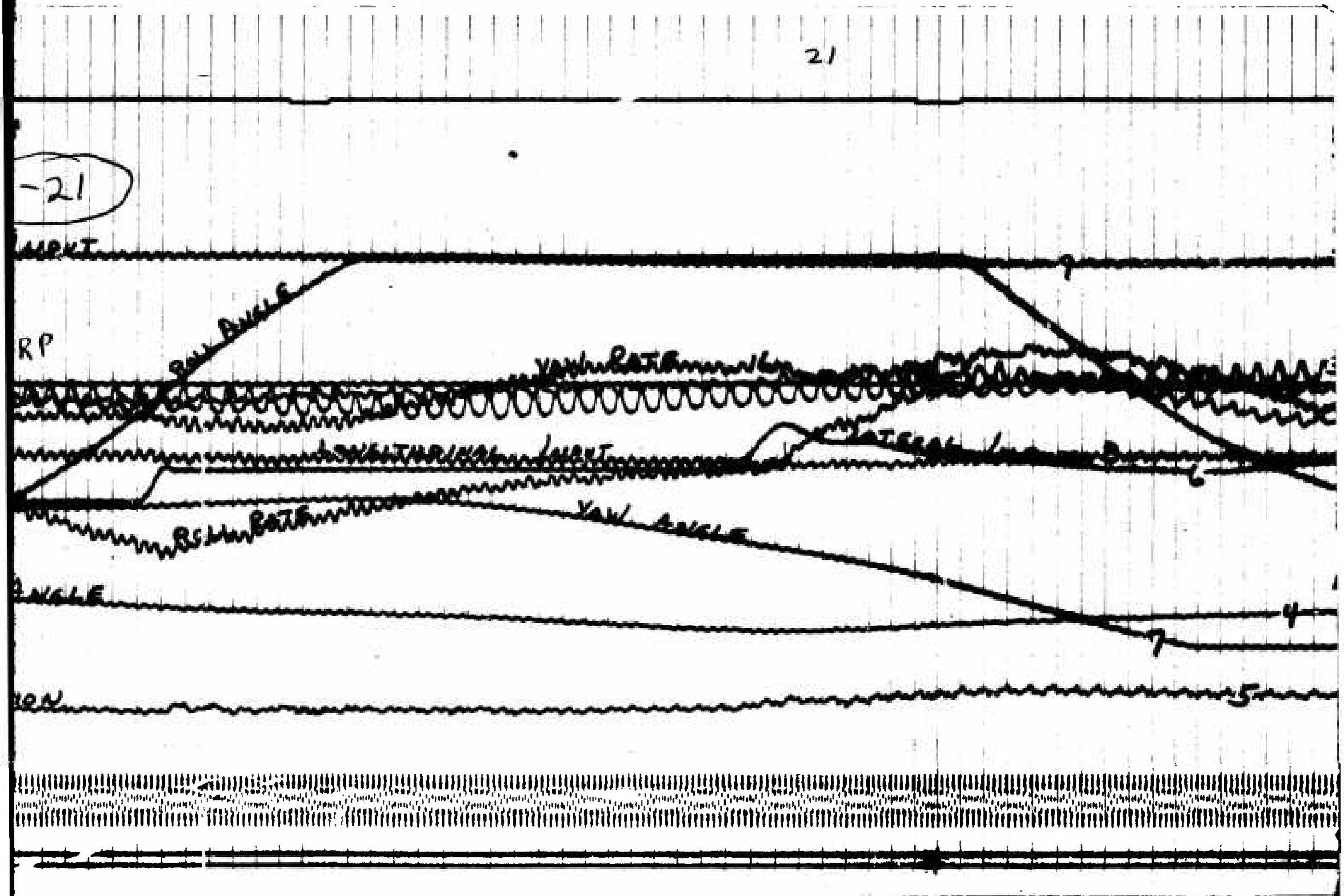


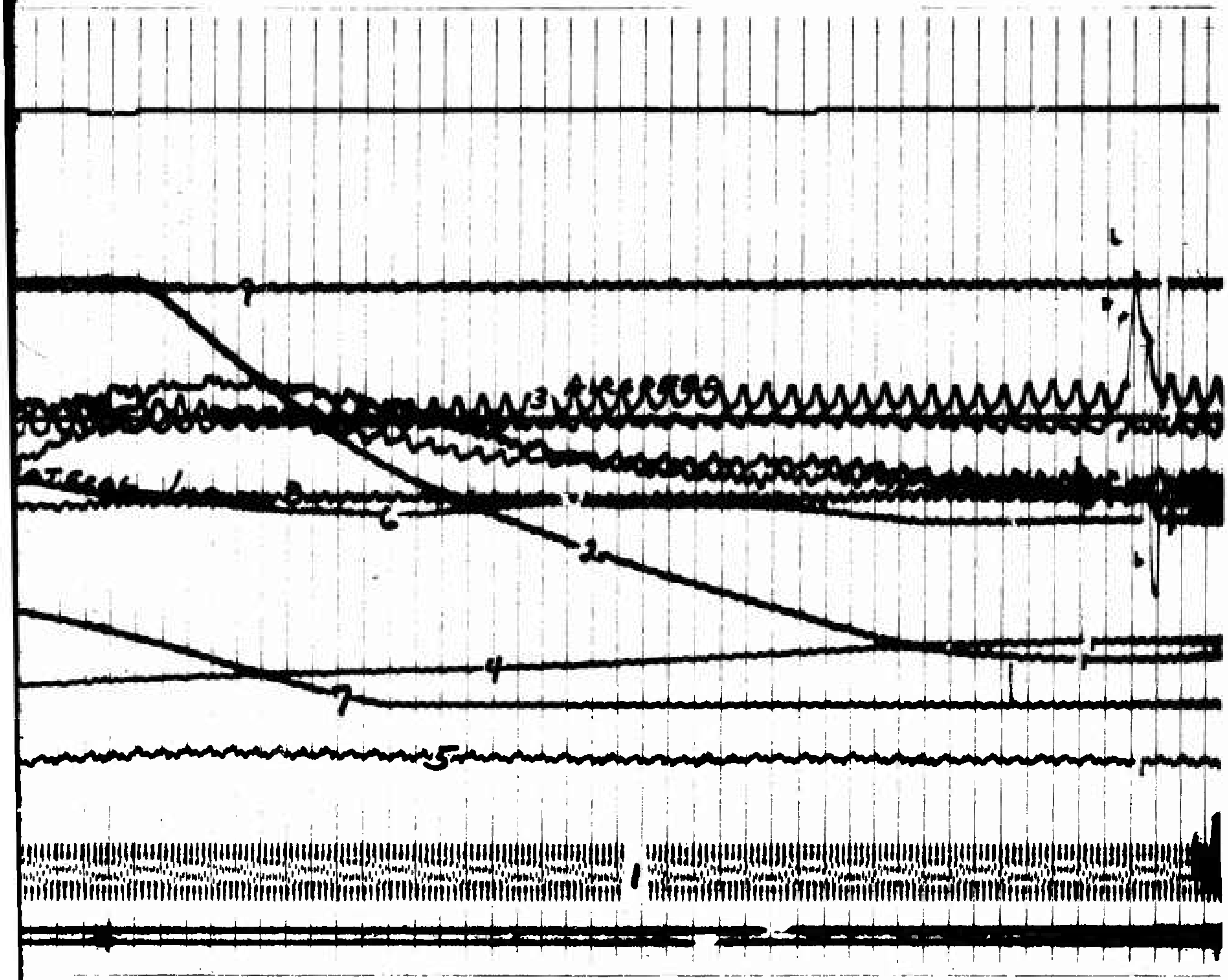
Figure 39. Right Lateral Pulse - 90 KIAS, 4000 Feet, Without Stabilizer Bar and SAS.

A



000 Feet,

B


7
3
6
2
4
7
5
1

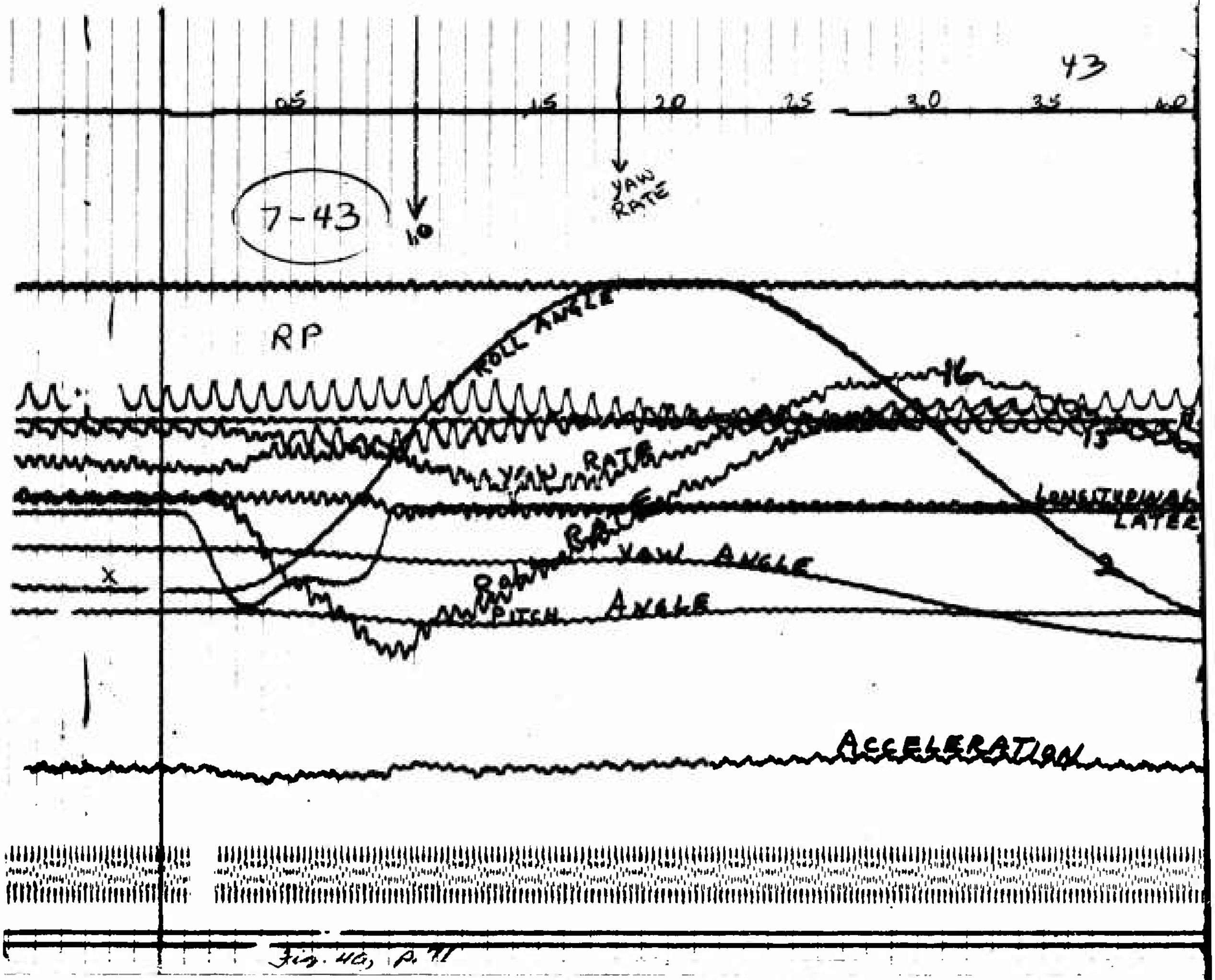
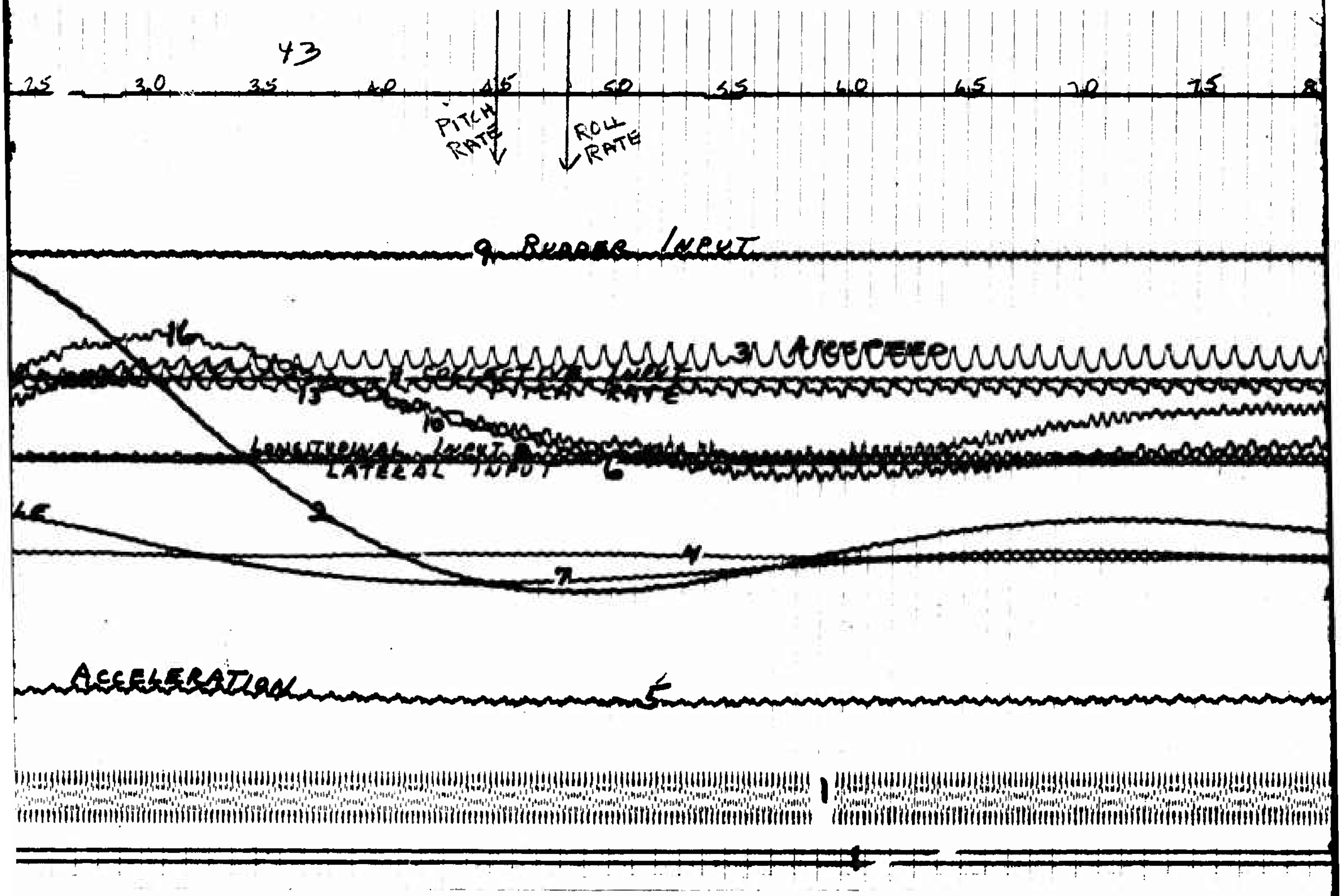
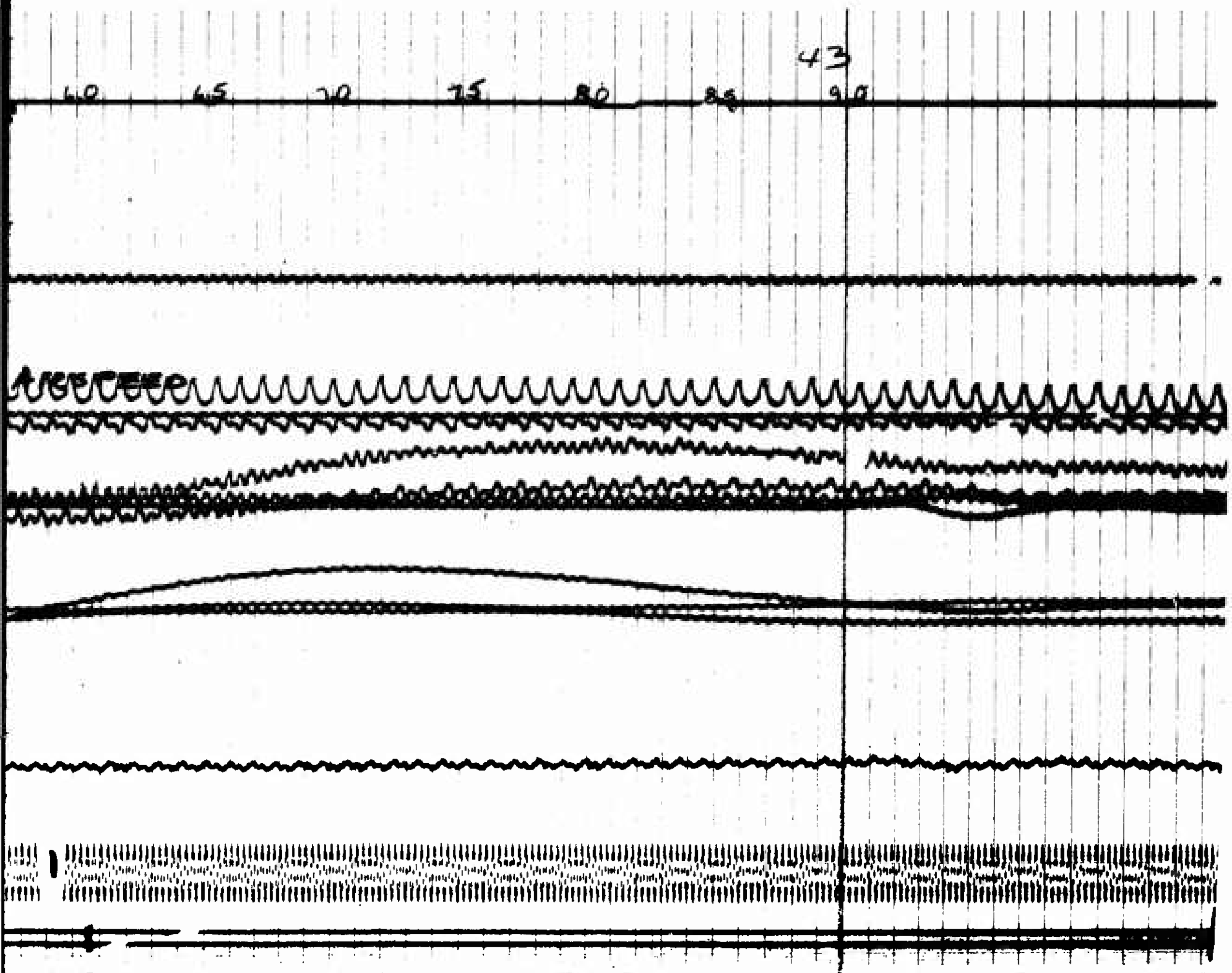


Figure 40. Right Lateral Pulse - 90 KIAS, 4000 Feet, With MSAS.

A


43
25
3.0
3.5
4.0
4.5
5.0
5.5
6.0
6.5
7.0
7.5
PITCH RATE
ROLL RATE
9 RUDDER INPUT
16
COLLECTIVE INPUT
PITCH RATE
13
10
LONGITUDINAL INPUT
LATERAL INPUT
6
2
4
7
ACCELERATION
5
1

B



C

APPENDIX II

PILOT EVALUATION REPORT

Prior to any experimental flight testing, the following should be accomplished:

1. Complete laboratory testing within an "aircraft" environment simulation. (The equipment should be tested under the conditions such as temperatures, vibrations, accelerations, and rates).

2. Complete system analysis of the equipment to be evaluated with particular emphasis directed at how any equipment limitations will affect the safety and functioning of the flight test program.

Comments pertaining specifically to the flight evaluation of the MSAS are as follows:

1. The yaw SAS was not functioning properly, which resulted in a degradation of yaw stability from that of the basic aircraft.

2. Pitch-roll coupling resulting from directional control inputs could have been the result of improper yaw SAS inputs.

3. Pitch stability provided by the MSAS during forward flight appeared to be improved over that provided by the stabilizer bar.

4. Lateral stability provided by the MSAS during forward flight was only slightly better than that available from the stabilizer bar-off configuration, and was appreciably degraded from that stability provided by the stabilizer bar.

5. Stability provided by the MSAS in hovering flight was not adequate and was severely degraded from the stability provided by the stabilizer bar (normal configuration). This was probably caused by the fact that the MSAS receives its corrective signals from the MSAS gyros mounted on the cargo compartment floor. This means that the aircraft must go through considerable attitude changes prior to MSAS corrective input. These attitude changes

and associated rates are sensed and perceived by the pilot, resulting in pilot imposed corrective input prior to MSAS inputs. Since the stabilizer bar (normal aircraft configuration) is directly connected to the rotor head, the stabilizing inputs from the stabilizer bar are made prior to perception by the pilot, hence more stability and less pilot inputs. In order for the MSAS to be effective during hovering flight, the gyros must be configured so that they can sense destabilizing disturbances and make stabilizing inputs prior to the perception of the initial aircraft disturbances by the pilot.

6. Extremely restrictive aircraft attitude limitations precluded a sound qualitative evaluation of the MSAS.

The MSAS, at its present stage of development, is not suitable for evaluating in the concept of a helicopter mechanical stability augmentation system.

Unclassified

Security Classification

DOCUMENT CONTROL DATA - R & D

(Security classification of title, body of abstract and indexing annotation must be entered when the overall report is classified)

1. ORIGINATING ACTIVITY (Corporate author) Eustis Directorate U.S. Army Air Mobility R&D Laboratory Fort Eustis, Virginia	2a. REPORT SECURITY CLASSIFICATION Unclassified 2b. GROUP

3. REPORT TITLE

FLIGHT TEST EVALUATION OF THE THREE-AXIS MECHANICAL STABILITY AUGMENTATION SYSTEM

4. DESCRIPTIVE NOTES (Type of report and inclusive dates)

5. AUTHOR(S) (First name, middle initial, last name)

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13. ABSTRACT

This note presents the results of flight tests conducted to evaluate a three-axis mechanical stability augmentation system (MSAS), known as "Dynagyro", on a UH-1 helicopter. The purpose of a stability augmentation system is to augment the stability and control characteristics of unstable or weakly stable aircraft so as to provide satisfactory flying qualities. The tests encompassed 9-1/2 flight hours and approximately 3 hours of ground and hangar testing. The MSAS included an entirely new concept: vortex valve fluidic servoactuators.

The magnitude of the installation and conversion procedure for installing the MSAS was relatively small.

The MSAS, as tested, did not perform as well in the helicopter as it did in the laboratory or as well as theory indicated it should.

The MSAS did not require a heat exchanger for fluid temperature control.

The engagement and disengagement transients were acceptable. Helicopter response was decreased significantly following small-amplitude step displacement of the flight controls. The MSAS was ineffective in improving lateral-directional damping.

The yaw SAS responded properly during autorotational entry; however, it functioned improperly during the remaining tests. Pilot acceptance of the MSAS was poor. The magnitude of the installation and conversion procedure for the test MSAS is not representative of the procedure for a production MSAS. The MSAS was not compatible with the operating environment of the UH-1H helicopter.

The improper functioning of the yaw SAS contributed to the ineffectiveness of the MSAS in improving lateral-directional damping. The yaw SAS provides insignificant yaw damping during autorotational entry. Improper MSAS functioning and helicopter attitude limitations contributed to the poor pilot acceptance of the MSAS.

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